NASA CR-66419

VOLUME III

SUBSTANTIATING TECHNICAL DATA

NORTHROP NORAIR

COPY NO 22

AIRCRAFT PRELIMINARY DESIGN REPORT V/STOL JET OPERATIONS RESEARCH AIRPLANE DESIGN STUDY - VOLUME III - SUBSTANTIATING TECHNICAL DATA

Prepared for

National Aeronautics and Space Administration
Langley Research Center
Langley Station
Hampton, Virginia 23365

(Contract NAS1-6777)

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ABSTRACT

This study is presented in three volumes:

Volume I – Airplane Specification for Northrop N-309 NASA V/STOL Jet Operations Research Airplane (NASA CR-66417)

Volume II – Airplane Specification for Northrop Modified T-39A NASA V/STOL Jet Operations Research Airplane (NASA CR-66418)

Volume III – Substantiating Technical Data (NASA CR-66419)

These reports provide a preliminary design description of a new and modified V/STOL jet operations research airplane. This preliminary design was conducted by Northrop Norair during Part III of Contract NAS1-6777, "Jet V/STOL Operations Research Airplane Design Study."

1.0 STUDY PART III CONFIGURATIONS

The final study configurations are presented on the following pages. In addition to the basic general arrangement drawings, the inboard profile, cockpit arrangement, and visibility diagrams are shown for the new (N-309) and modified (T-39A) vehicles.

N-309 Drawings:

AD-4486	General arrangement
AD-4513	Inboard profile
AD-4433B	Front cockpit arrangement
AD-4434B	Rear cockpit arrangement
Diagram	Visibility (front and rear seat)
General Data	

Modified T-39A Drawings:

AD-4448	General arrangement
AD-4512	Inboard profile
AD-4517	Cockpit arrangement
Diagram	Visibility (right and left hand seat)
General Data	

N-309 AIRPLANE

The N-309 as shown in Figure 1-1, drawing AD-4486 (A CHG.), has seven lift engines in the fuselage and two lift/cruise engines mounted in nacelles. The position of the aft lift engine provides adjacent space to divert the lift/cruise exhaust through the body for the composite lift mode. This feature provides a versatile bay for installation of an additional lift engine by removing the lift/cruise diverted exhaust system and remounting the centerline lift engine to make room for the second lift engine required for the direct lift mode of flight.

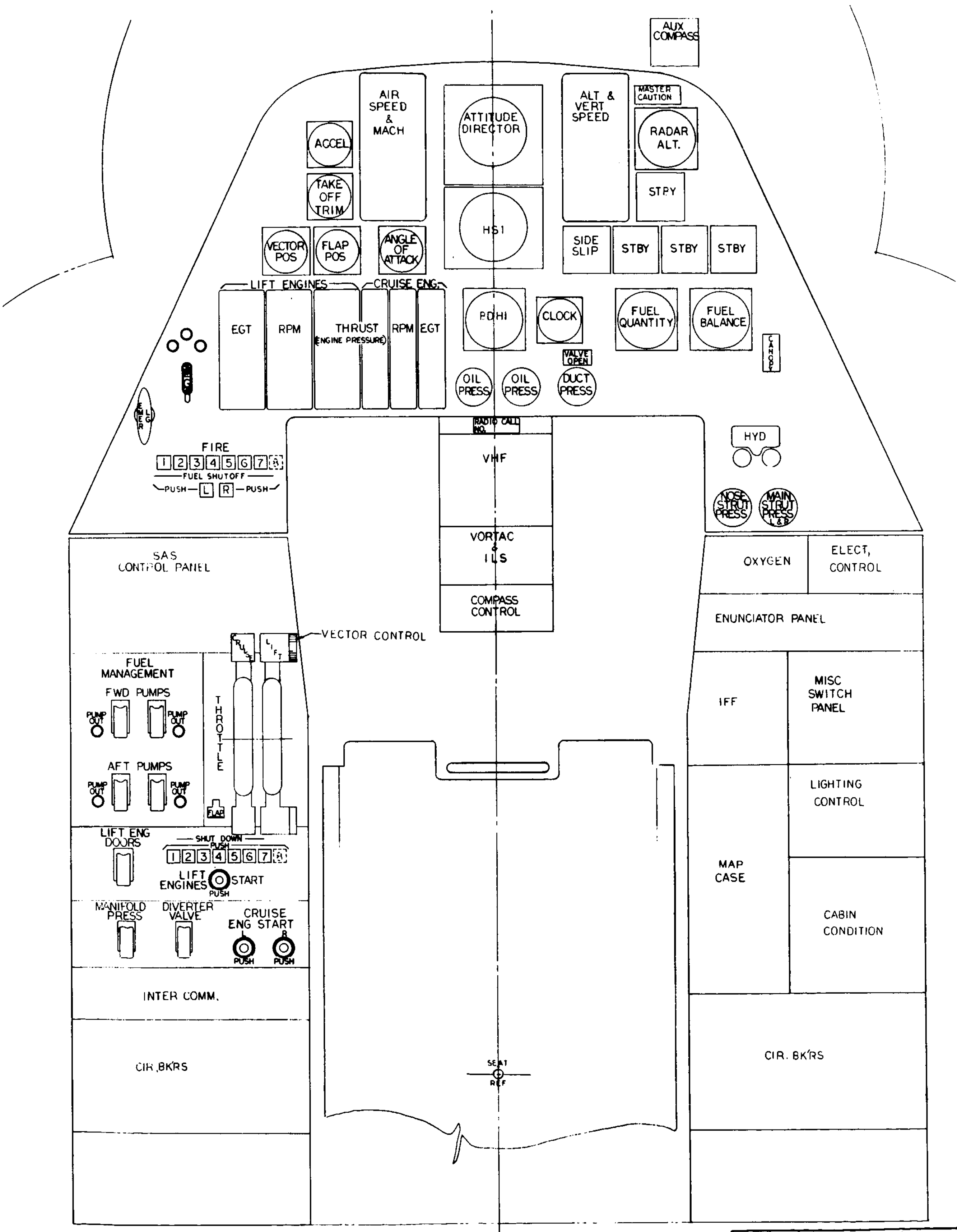
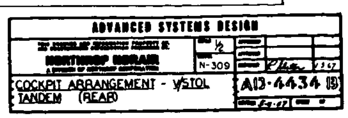

AUX COMPASS
AIR SPEED & MACH
ATTITUDE DIRECTOR
ALT & VERT SPEED
MASTER CAUTION
RADAR ALT.
ACCEL
TAKE OFF TRIM
STBY
HSI
VECTOR POS
FLAP POS
ANGLE OF ATTACK
SIDE SLIP
STBY
STBY
STBY
LIFT ENGINES
CRUISE ENG
EGT
RPM
THRUST (ENGINE PRESSURE)
RPM
EGT
PDHI
CLOCK
FUEL QUANTITY
FUEL BALANCE
CANOPY
VALVE OPEN
OIL PRESS
OIL PRESS
DUCT PRESS
EMER
FIRE
1 2 3 4 5 6 7 8
FUEL SHUTOFF
PUSH L R PUSH
RADIO CALL NO.
VHF
HYD
NOSE STRUT PRESS
MAIN STRUT PRESS L&R
VORTAC & ILS
COMPASS CONTROL
SAS CONTROL PANEL
OXYGEN
ELECT. CONTROL
ENUNCIATOR PANEL
VECTOR CONTROL
CRUISE
LIFT
FUEL MANAGEMENT
FWD PUMPS
PUMP OUT
PUMP OUT
THROTTLE
AFT PUMPS
PUMP OUT
PUMP OUT
FLAP
IFF
MISC SWITCH PANEL
LIGHTING CONTROL
LIFT ENG DOORS
SHUT DOWN PUSH
1 2 3 4 5 6 7 8
LIFT ENGINES START
PUSH
MAP CASE
MANIFOLD PRESS
DIVERTER VALVE
CRUISE ENG START
L R
PUSH
PUSH
CABIN CONDITION
INTER COMM.
CIR.BK'RS
SEAT REF
CIR. BK'RS

FIGURE 1-4


ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
1/2
N-309
COCKPIT ARRANGEMENT - V/STOL TANDEM (REAR)
AD-4434

AITOFF'S EQUAL AREA PROJECTION OF THE SPHERE

RADIUS OF PROJECTED SPHERE EQUALS ONE DECIMETER

VTOL TANDEM COCKPIT VISIBILITY (N-309)

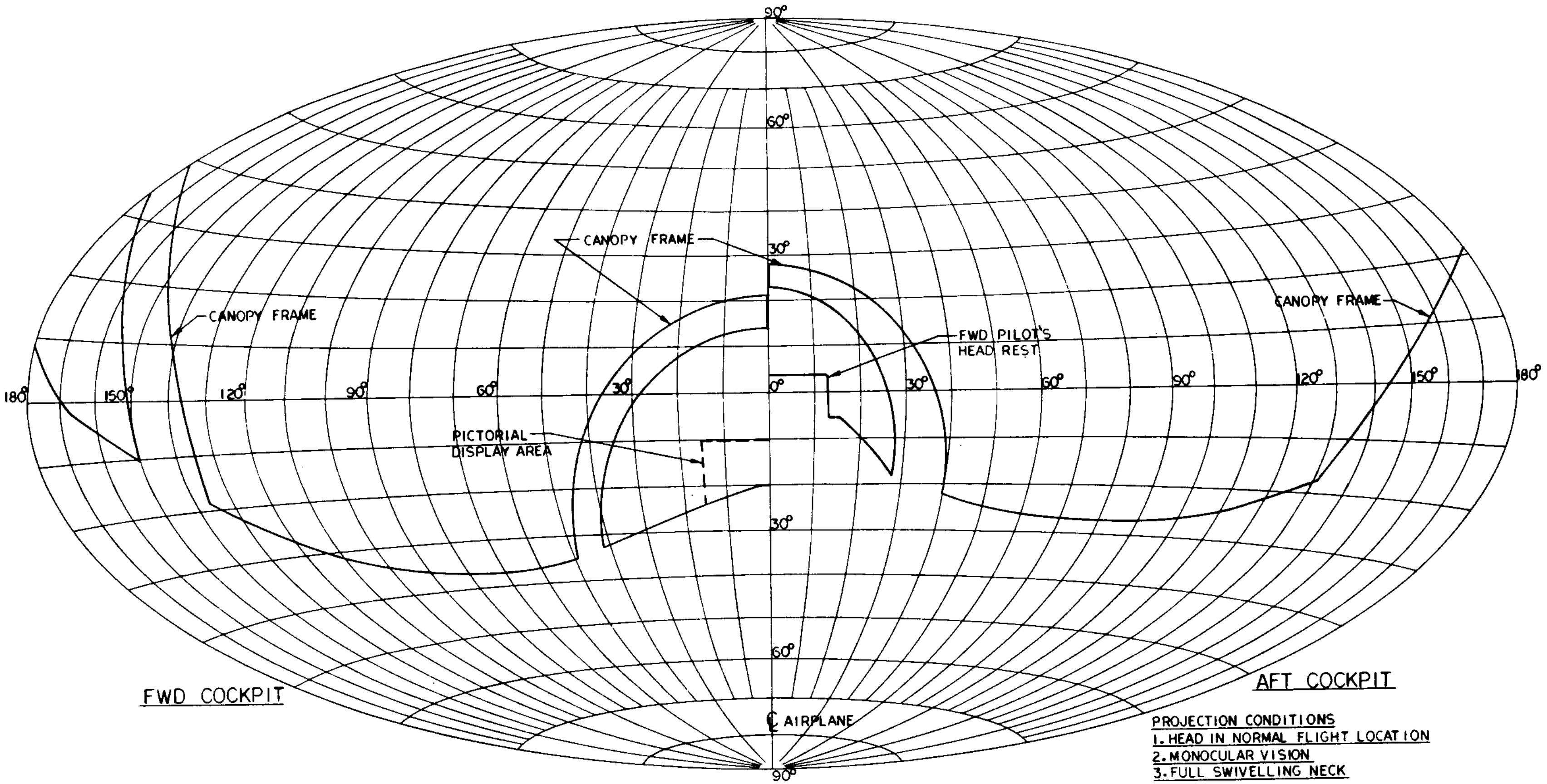


FIGURE 1-5. AIRPLANE IN STATIC GROUND POSITION

GENERAL DATA - N-309

AREAS

Wing Area Total (Theoretical) Including Aileron and Flap	210 Sq. Ft.
Wing Trailing Edge Flap Area	23.2 Sq. Ft.
Wing Leading Edge Flap Area	23.2 Sq. Ft.
Aileron Area Aft of Hinge Line	8.8 Sq. Ft.
Horizontal Tail Area Total	66.6 Sq. Ft.
Exposed	45 Sq. Ft.
Vertical Tail Area Total (Theoretical, not Including Dorsal)	48 Sq. Ft.
Rudder Area Aft of Hinge Line	10.6 Sq. Ft.

DIMENSIONS AND GENERAL DATA

Wing	
Span (Maximum)	36.8 Ft.
Theoretical	35.5 Ft.
Chord	
At Root	101.5 In.
At Construction Tip	40.6 In.
Mean Aerodynamic	75.3 In.
Airfoil Section	
At Root	NACA64A013
At Tip	NACA64A012
Incidence Degrees	0^{o}
Sweepback at 25% Chord Degrees	20^{o}
Dihedral Degrees	0^{o}
Aspect Ratio	6
Taper Ratio	.40
Ailerons	
Span	4.25 Ft.
Chord/Wing Chord	25%

Flap Trailing Edge	
Span	8 Ft.
Chord/Wing Chord	25%
Flap Leading Edge	
Span	11.3 Ft.
Chord/Wing Chord	20%
Horizontal Tail	
Span	15.7 Ft.
Chord	
Root	78.5 In.
Tip	24 In.
Mean Aerodynamic	48 In.
Airfoil Section	NACA64A010
Sweepback at 25% Chord Degrees	27°
Dihedral Degrees	$-10^{\circ}30'$
Aspect Ratio	3.7
Taper Ratio	.30
Vertical Tail	
Span	7.6 Ft.
Chord	
Root	117 In.
Tip	35 In.
Mean Aerodynamic	83 In.
Airfoil Section	NACA64A010
Sweepback at 25% Chord Degrees	40°
Aspect Ratio	1.2
Taper Ratio	.30

GENERAL

Height Above Ground Over Highest Fixed Part of Aircraft	
All Wheels in Contact	15 Ft. 4 In.
Length, Maximum	55 Ft. 7 In.
Distance From Wing Mac Quarter Chord Point to Horizontal Tail Mac Quarter Chord Point	180 In.

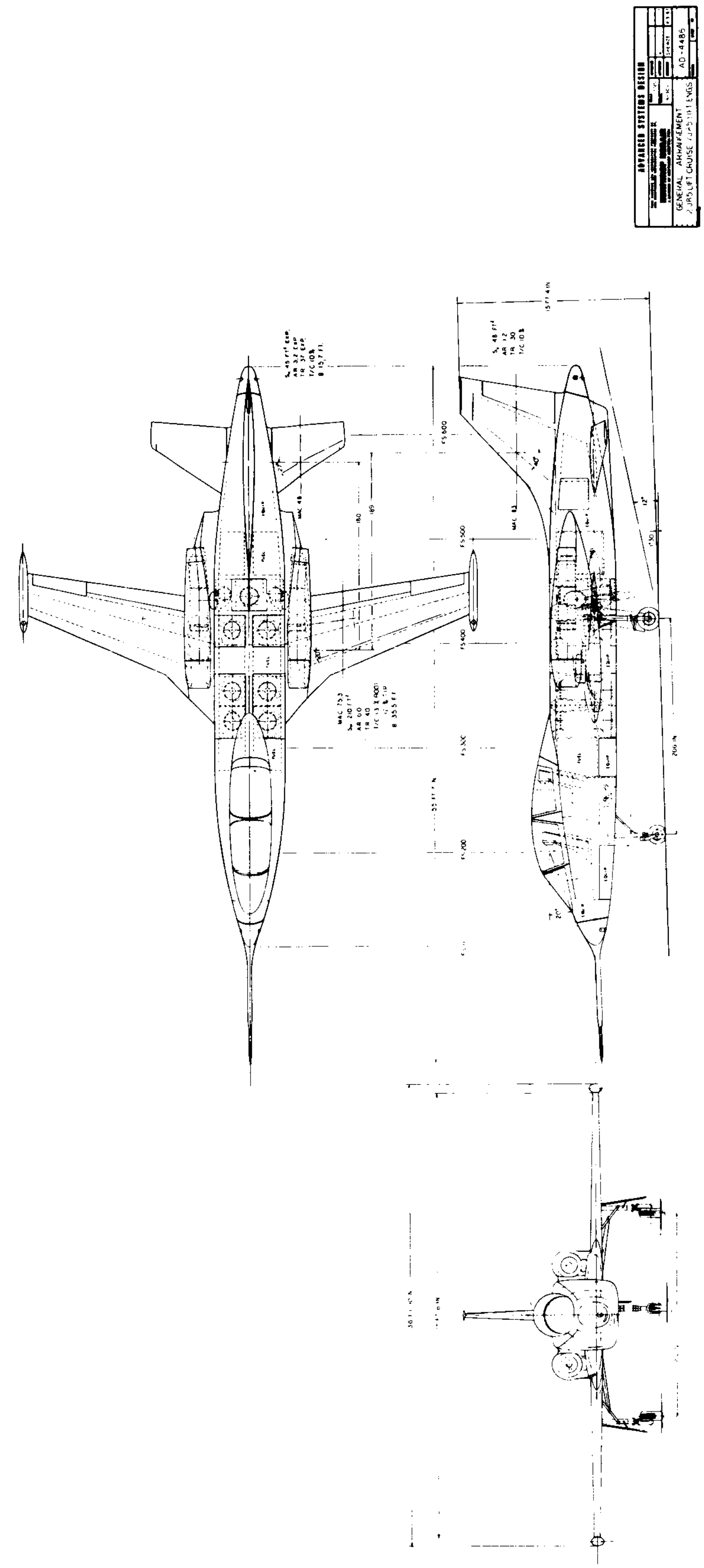


FIGURE 1-1

N-309 INBOARD PROFILE

The N-309 inboard profile drawing, AD-4513 (Figure 1-2), presents the internal arrangement for all major components including the crew stations. The position of the engines, both lift/cruise and lift, the fuel system, landing gear, reaction control and primary flight control system, equipment space and subsystems are indicated in the detailed profile and cross-sections of the vehicle. This layout establishes the fact that sufficient fuselage volume is available to meet systems installation requirements.

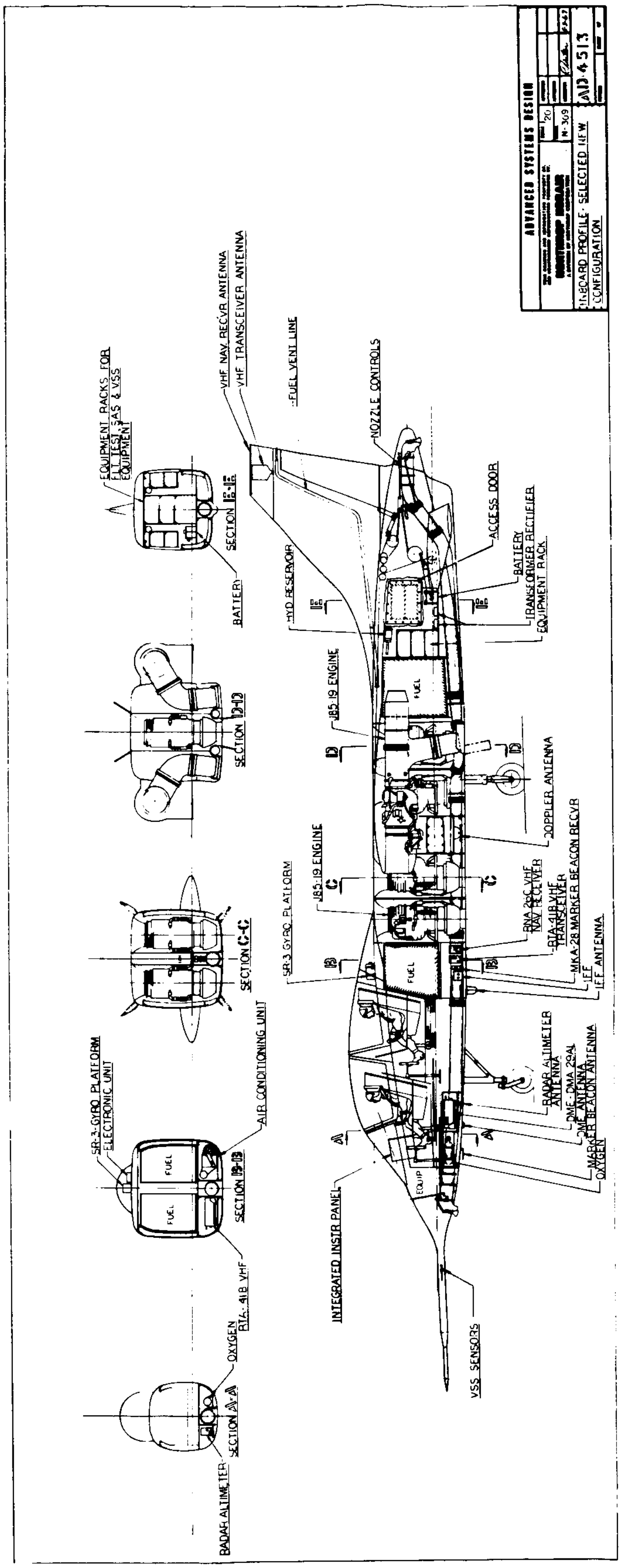

ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
INBOARD PROFILE - SELECTED HFW CONFIGURATION
AD 4513
RADAR ALTIMETER
SECTION A-A
OXYGEN
RTA-41B VHF
SECTION B-B
AIR CONDITIONING UNIT
SR-3 GYRO PLATFORM
ELECTRONIC UNIT
FUEL
SECTION C-C
SECTION D-D
BATTERY
SECTION E-E
EQUIPMENT RACKS FOR FLT TEST, SAS & VSS EQUIPMENT
VHF NAV RECVR ANTENNA
VHF TRANSCEIVER ANTENNA
FUEL VENT LINE
NOZZLE CONTROLS
ACCESS DOOR
BATTERY
TRANSFORMER RECTIFIER
EQUIPMENT RACK
HYD RESERVOIR
J85-19 ENGINE
DOPPLER ANTENNA
J85-19 ENGINE
SR-3 GYRO PLATFORM
RNA-26C VHF NAV RECEIVER
RTA-41B VHF TRANSCEIVER
MKA-28 MARKER BEACON RECVR
IFF
IFF ANTENNA
RADAR ALTIMETER ANTENNA
DME-DMA 29AL
DME ANTENNA
MARKER BEACON ANTENNA
OXYGEN
EQUIP
INTEGRATED INSTR PANEL
VSS SENSORS

FIGURE 1-2.

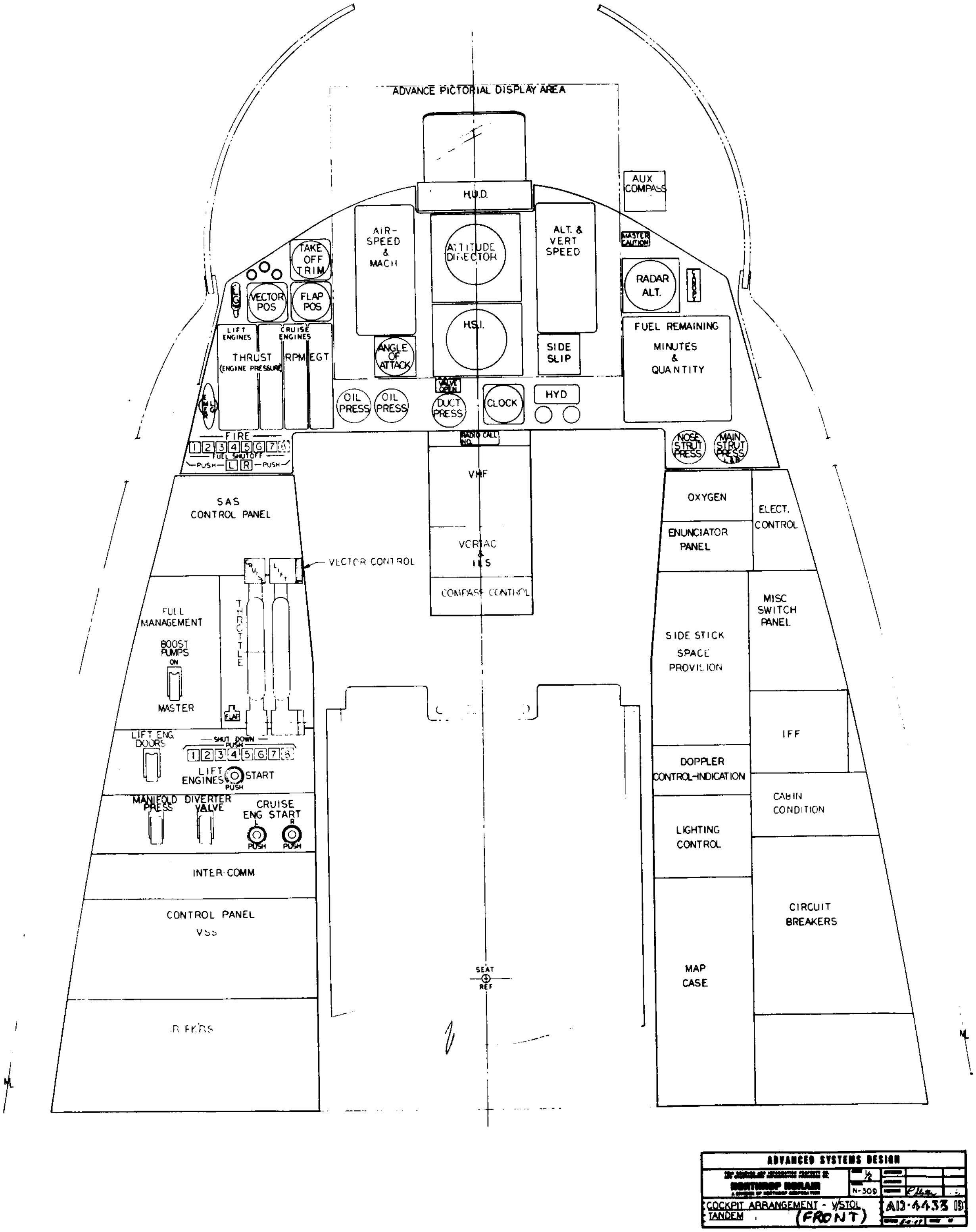

ADVANCE PICTORIAL DISPLAY AREA
H.U.D.
AUX COMPASS
AIR-SPEED & MACH
ATTITUDE DIRECTOR
ALT. & VERT SPEED
TAKE OFF TRIM
VECTOR POS
FLAP POS
RADAR ALT.
H.S.I.
LIFT ENGINES
CRUISE ENGINES
THRUST (ENGINE PRESSURE)
RPM
EGT
ANGLE OF ATTACK
SIDE SLIP
FUEL REMAINING
MINUTES & QUANTITY
OIL PRESS
OIL PRESS
DUCT PRESS
CLOCK
HYD
FIRE
FUEL SHUTOFF
PUSH
PUSH
NOSE STRUT PRESS
MAIN STRUT PRESS
VHF
SAS CONTROL PANEL
OXYGEN
ELECT. CONTROL
ENUNCIATOR PANEL
VECTOR CONTROL
VORTAC
ILS
COMPASS CONTROL
FUEL MANAGEMENT
BOOST PUMPS
ON
MASTER
THROTTLE
MISC SWITCH PANEL
SIDE STICK SPACE PROVISION
IFF
LIFT ENG DOORS
SHUT DOWN
PUSH
LIFT ENGINES
START
PUSH
DOPPLER CONTROL-INDICATION
CABIN CONDITION
MANIFOLD PRESS
DIVERTER VALVE
CRUISE ENG START
PUSH
PUSH
LIGHTING CONTROL
INTER-COMM
CIRCUIT BREAKERS
CONTROL PANEL
VSS
SEAT REF
MAP CASE
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
N-309
COCKPIT ARRANGEMENT - V/STOL TANDEM
(FRONT)
AD-4433

FIGURE 1-3

Distance From Wing Mac Quarter Chord Point to Vertical Tail Mac Quarter Chord Point	189 In.
Angle Between Reference Line and Wing Zero Lift Line (All Surfaces Neutral)	0^{o}
Ground Angle, All Wheels Contact, Static	$1^{o}30'$
Max. Tail Down Static	12^{o}
Wheel Size (Rim Diameter)	
Main Wheel	11. In.
Nose Wheel	8. In.
Tire Size	
Main Wheel	22 x 8.5-11 Type VIII 14 Ply Rating
Nose Wheel	18 x 5.5 Type VII 8 Ply Rating
Tread of Main Wheels	194 In.
Wheel Base	206 In.
Vertical Travel of Axle From Extended to Fully Compressed Position	
Main Landing Gear	19.25 In.
Nose Landing Gear	20 In.

CONTROL MOVEMENT AND CORRESPONDING CONTROL SURFACE MOVEMENTS

Nominal control surface and control movement on each side of neutral position for full movements as limited by stops shall be as follows.

Rudder	$\pm 25^{o}$
Rudder Pedals	± 3.5 Inches Forward and Aft
Horizontal Stabilizer	25^{o} up 15^{o} down
Horizontal Stabilizer Control	6 In. Aft 4 In. Forward
Ailerons	20^{o} up 20^{o} down
Aileron Control	4.25 In. Right or Left
Wing Flap	
Trailing Edge	40^{o} Total
Leading Edge	25^{o} Total

MODIFIED T-39A AIRPLANE

The selected arrangement for the Mod T-39A configuration, as shown in Figure 1-6 (Drawing AD-4448), has an extended fuselage to facilitate the installation of eight lift engines for composite flight or ten lift engines for flight in a direct lift mode. The lift/cruise engines used in the composite mode are mounted in nacelles. However, the exhaust is directed down through the nacelle, external to the fuselage. A protective shield is placed about the main gear tire due to the lift engine exhaust temperatures. The reaction control system consists of two ducts external to the body to efficiently utilize the space available in the fuselage to meet other equipment requirements.

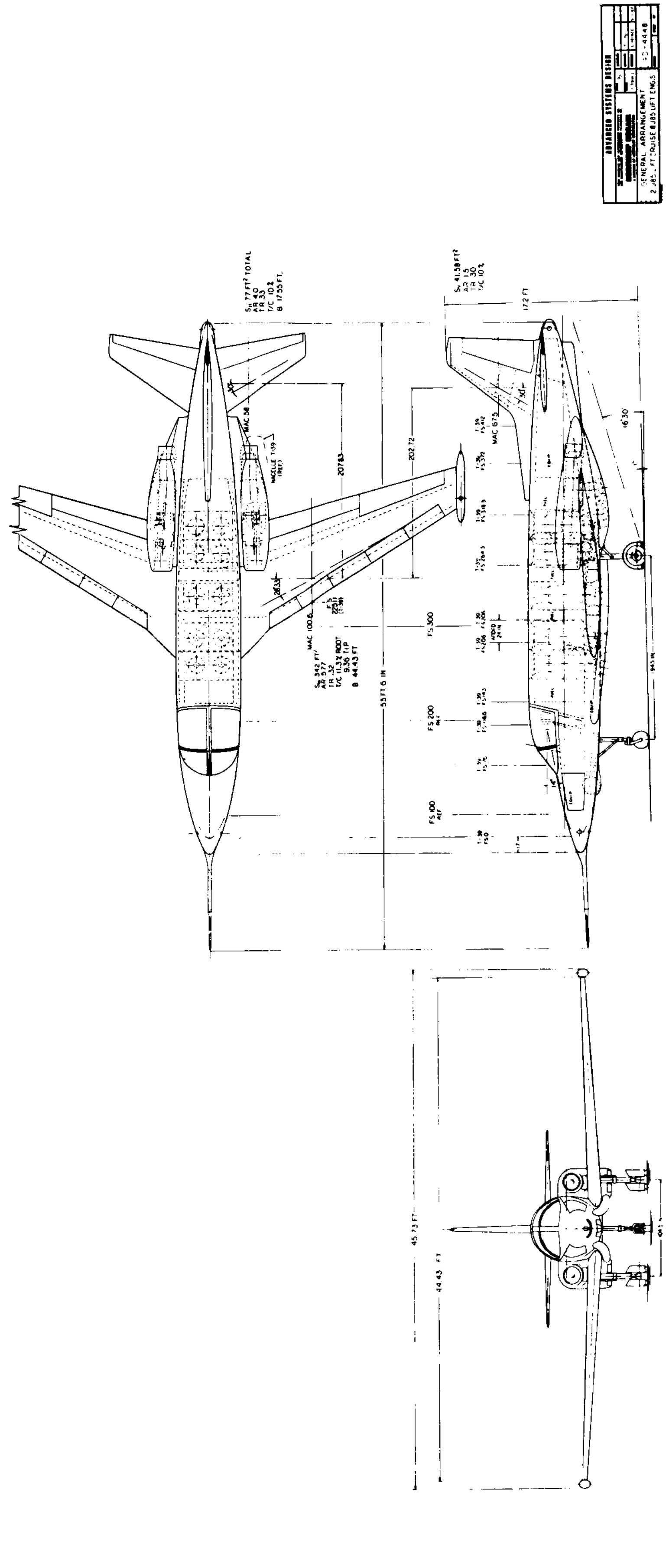


FIGURE 1-6

MOD T-39A INBOARD PROFILE

The internal arrangement of the modified T-39A is presented in the inboard profile drawing AD-4512 (Figure 1-7). The original T-39A airplane is altered to develop the required internal volume for lift engines, equipment and subsystems. The lift cruise engine nacelles are relocated over the trailing edge of the wing and are also used to stow the A-4E main landing gear. The original side-by-side arrangement in the cockpit is the same, however due to the installation of ejection seats changes are required to the canopy and fuselage structure. New instrument panels and consoles are also necessary. The fuel system and reaction control plumbing are shown in the profile and the section. None of the original T-39A subsystems can be retained without modification.

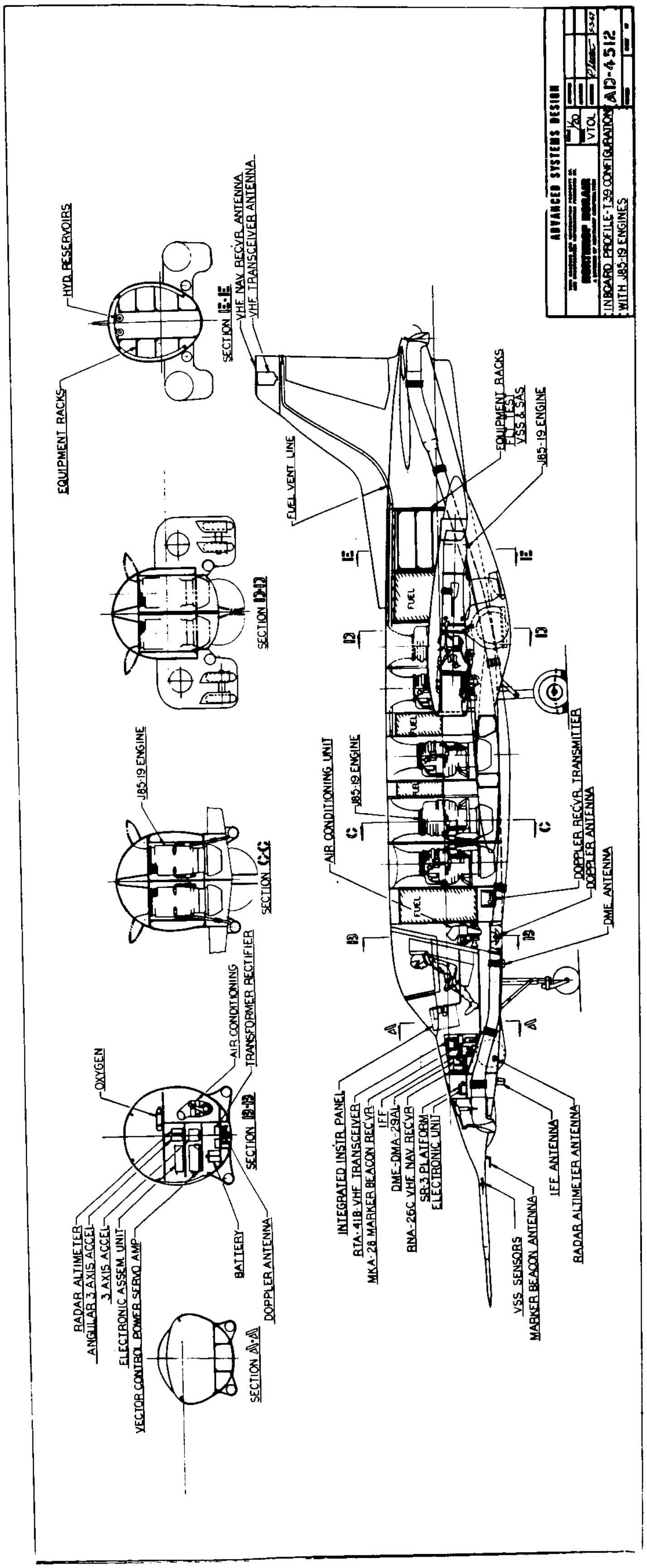
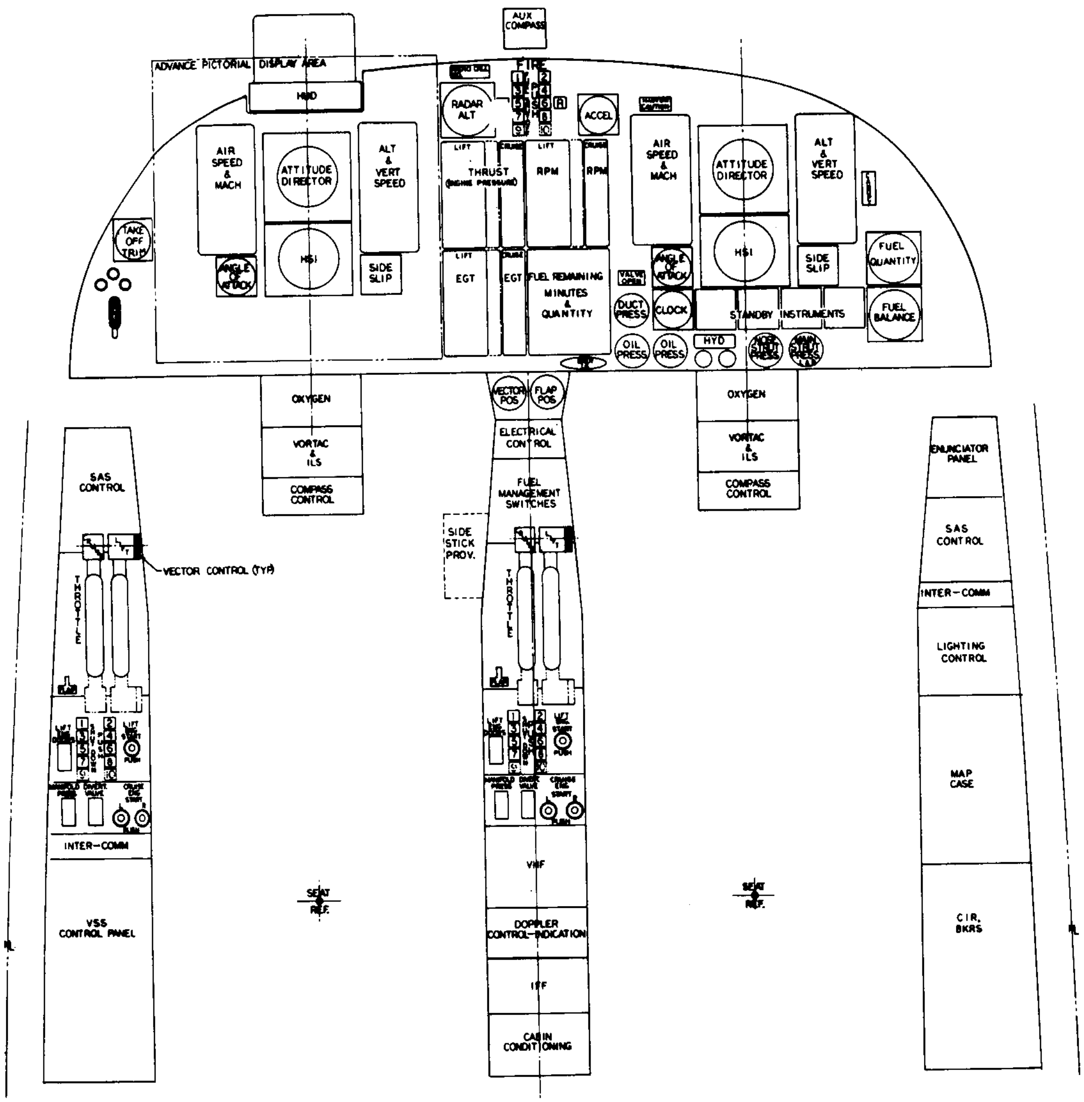

HYD RESERVOIRS
EQUIPMENT RACKS
SECTION E-E
VHF NAV RECVR ANTENNA
VHF TRANSCEIVER ANTENNA
SECTION D-D
J85-19 ENGINE
SECTION C-C
AIR CONDITIONING
TRANSFORMER RECTIFIER
OXYGEN
SECTION B-B
RADAR ALTIMETER
ANGULAR 3 AXIS ACCEL
3 AXIS ACCEL
ELECTRONIC ASSEM UNIT
VECTOR CONTROL POWER SERVO AMP
BATTERY
DOPPLER ANTENNA
SECTION A-A
FUEL VENT LINE
EQUIPMENT RACKS
FLT TEST
VSS & SAS
J85-19 ENGINE
AIR CONDITIONING UNIT
J85-19 ENGINE
FUEL
DOPPLER RECVR TRANSMITTER
DOPPLER ANTENNA
DME ANTENNA
INTEGRATED INSTR PANEL
RTA-41B VHF TRANSCEIVER
MKA-28 MARKER BEACON RECVR
IFF
DME-DMA-29AL
RNA-26C VHF NAV RECVR
SB-3 PLATFORM
ELECTRONIC UNIT
VSS SENSORS
MARKER BEACON ANTENNA
IFF ANTENNA
RADAR ALTIMETER ANTENNA
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
VTOL
INBOARD PROFILE-T-39 CONFIGURATION
WITH J85-19 ENGINES
AD-4512

FIGURE 1-7



ADVANCED SYSTEMS DESIGN	
NORTHROP NORAIR	1/2 V/STOL GARRETT
COCKPIT ARRANGEMENT-V/STOL SIDE BY SIDE, T-39	AD-4517

FIGURE 1-8.

AITOFF'S EQUAL AREA PROJECTION OF THE SPHERE

RADIUS OF PROJECTED SPHERE EQUALS ONE DECIMETER

MOD. T-39 COCKPIT VISIBILITY

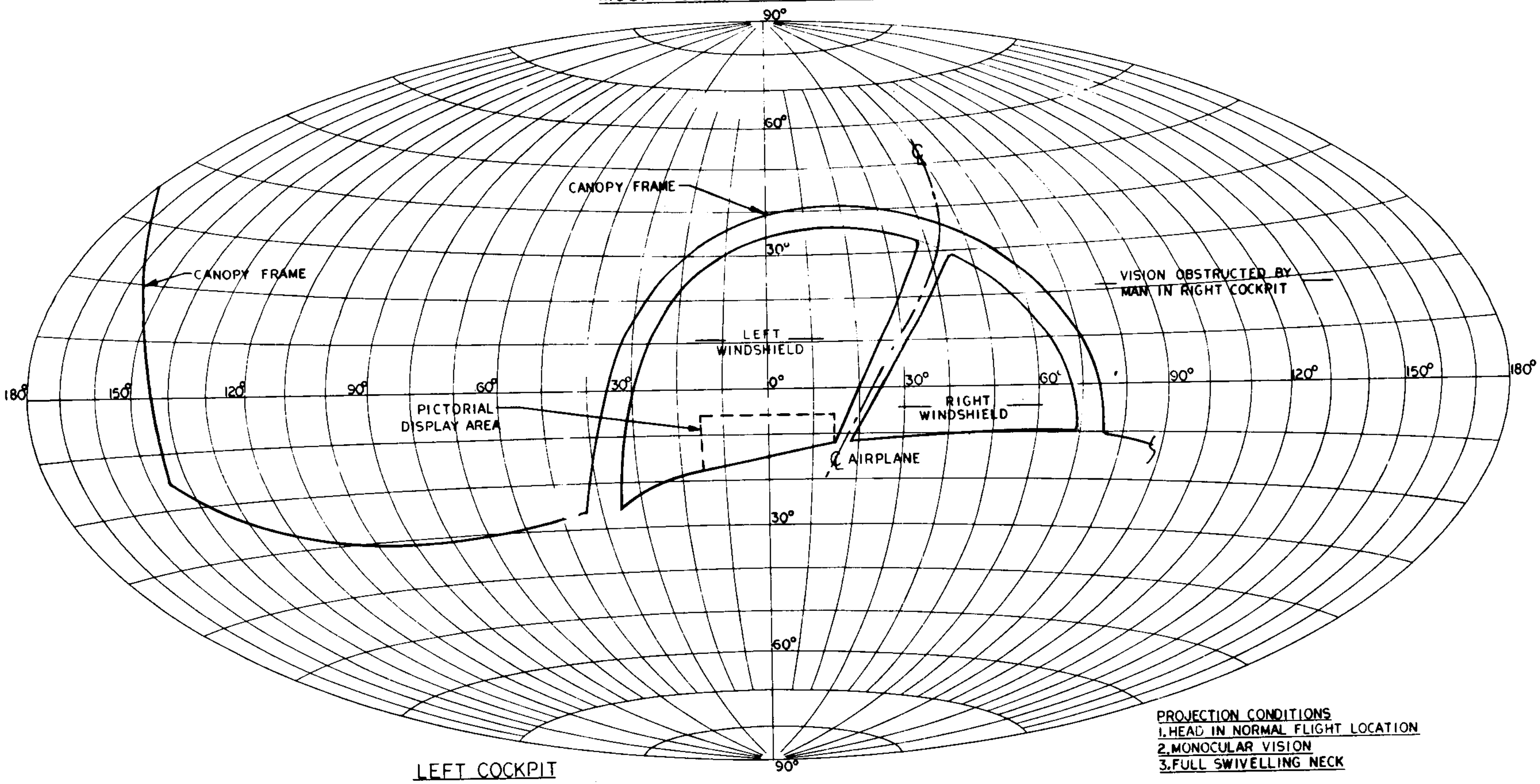


FIGURE 1-9. LEFT COCKPIT - AIRPLANE IN STATIC GROUND POSITION

GENERAL DATA - MOD. T-39A

AREAS

Wing Area Total (Theoretical) Including Aileron and Flap Area	342 Sq. Ft.
Wing Flap Total	39.5 Sq. Ft.
Wing Slats Total	36.34 Sq. Ft.
Aileron Area Aft of Hinge Line Total	16.42 Sq. Ft.
Horizontal Tail Area, Total	77 Sq. Ft.
Vertical Tail Area Total Not Including Dorsal	41.58 Sq. Ft.
Rudder, Aft of Hinge Line	8.95 Sq. Ft.

DIMENSIONS AND GENERAL DATA

Wing	
Span Maximum	45.73 Ft.
Theoretical	44.43 Ft.
Chord	
At Root	139.86 In.
At Construction Tip	44.91 In.
Mean Aerodynamic	100.6 In.
Airfoil Section	
At Root	64A Series (NAA Mod.) 11.3 Pct
At Tip	64A Series (NAA Mod.) 9.36 Pct
Incidence	
At Root	0 Degree
At Tip	$-2^{o}54'$
Sweepback at 25% Chord	$28^{o}33'$
Dihedral	$3^{o}09'$
Aspect Ratio	5.77
Taper Ratio	.32
Ailerons	
Span (each)	87.28 In.
Chord/Wing Chord	.20

Slats	
Span (each)	178.2 In.
Chord (mean)	14.72 In.
Flaps	
Span (each)	98. In.
Chord/Wing Chord	.281
Horizontal Tail	
Span	17.55 Ft.
Chord	
At Root	81.22 In.
At Tip	24.37 In.
Mean Aerodynamic	58 In.
Airfoil Section	NACA64A010 (Mod)
Sweepback at 25% Chord	30^o
Dihedral	0^o
Aspect Ratio	4.0
Taper Ratio	.33
Vertical Tail	
Span	99 In.
Chord	
Root	100 In.
Tip	29 In.
Mean Aerodynamic	67.5
Airfoil Section	NACA63A010 Mod.
Sweepback	30^o
Aspect Ratio	1.5
Taper Ratio	.30

GENERAL

Height Above Ground Over Highest Fixed Part of Aircraft	17.2 Ft.
Length Maximum	55 Ft. 6 In.
Distance From Wing Mac Quarter Chord Point to Horizontal Tail Quarter Chord	207.8 In.
Distance From Wing Mac Quarter Chord Point to Vertical Tail Quarter Chord	202.7 In.

Ground Angle, All Wheels Contact Static	1^{o}
Max Tail Down Static	$16^{o}30'$
Wheel Size (Rim Diameter)	
Main Wheel	12 In.
Nose Wheel	8 In.
Tire Size	
Main Wheel	20 x 4.4 Type VII
Nose Wheel	18 x 5.5 Type VII
Tread of Main Wheels	104.5 In.
Wheel Base	194.5 In.
Vertical Travel of Axle From Extended to Fully Compress Position	
Main Landing Gear	14 In.
Nose Landing Gear	20 In.

CONTROL MOVEMENT AND CORRESPONDING CONTROL SURFACE MOVEMENTS

Nominal control surface and control movements on each side of neutral position for full movements as limited by stops shall be as follows.

Rudder	$\pm 25^{o}$
Rudder Pedals	± 3.5 Inches Forward and Aft
Horizontal Stabilizer	25^{o} up 15^{o} down
Horizontal Stabilizer Control	6 In. Aft, 4 In. Forward
Aileron	$\pm 16^{o}$
Aileron Control	4.25 In. Right or Left
Wing Slats	
Wing Flap	25^{o}

HUMAN FACTORS

A Human Factors review of the desired five minute limit for preflight checkout procedures has been performed on the N-309 and T-39A modified configurations. The study results indicate that the five minute limit for preflight checkout is attainable, provided that the following tasks are performed first:

Aircraft Form 1 checked; aircraft exterior visual inspection completed; all ground safety pins installed; pilot installed in the aircraft, with parachute straps and seat restraint harness fastened; oxygen hose and communications line connected; seat and rudder pedals adjusted; ground start cart connected and operating.

The above qualifications are required due to the unpredictable nature of the pilot's findings in the visual preflight inspection, and due to the differences in time to enter the cockpit and prepare for flight.

A preliminary task-timeline analysis of pilot tasks in the start and pre-liftoff sequence is shown on page 1-22. Times given for task performance are estimates, based on analyses of similar tasks in fixed wing aircraft, and performed by experienced pilots thoroughly familiar with the cockpit.

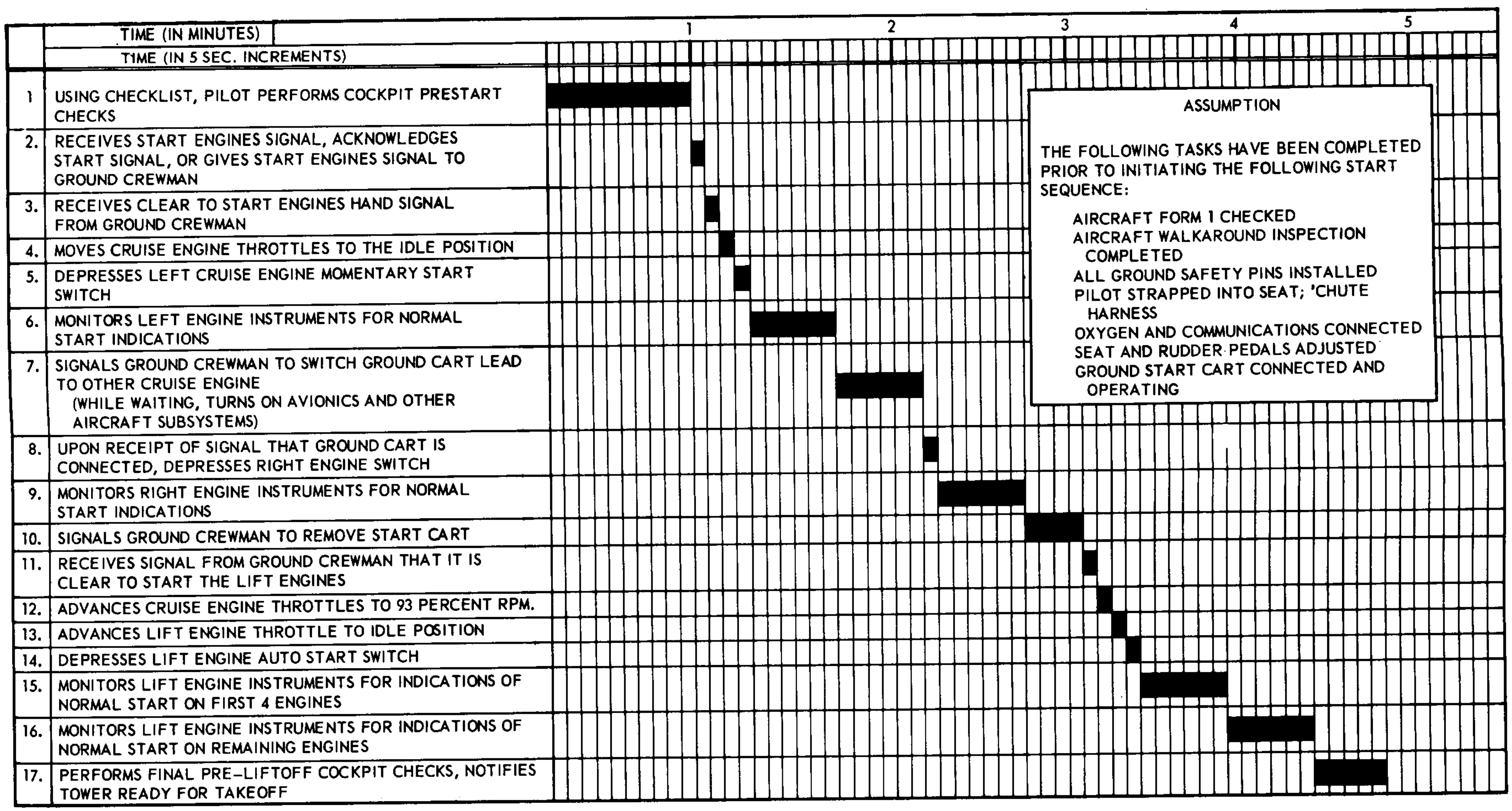


FIGURE 1-10. TASK-TIMELINE FOR START SEQUENCE

2.0 AERODYNAMICS

2.1 JET INDUCED INTERFERENCE EFFECTS

2.1.1 Hover Interference

The loss in lift due to low pressures induced by the jet exhaust along the lower planform surfaces is a function of several parameters associated with aircraft geometry and engine characteristics. Analysis of pertinent reports on lift loss in hover (References 2-1 to 2-5) has shown that lift loss is dependent upon the following criteria: (1) the total planform to jet exit area ratio; (2) the number of jet exits; (3) planform configuration; (4) the height of the wing and the height of the jet exit above the ground; (5) jet decay characteristics; and (6) jet pressure ratio. Even though the jet decay characteristics on the J-85-19 engine were unknown, sufficient data requirements were available to estimate the magnitude of lift loss for each configuration.

The method of approach which was followed assumed a test model configuration closest to the study configuration. Interference lift over thrust $\frac{\Delta L}{T}$, as a function of h/D_E, the ratio of height above ground to equivalent diameter $\left(\sqrt{\frac{4 \text{ x Total Nozzle Area}}{\pi}}\right)$ for the N-309 configuration is presented in Figure 2-1. Since the number of jet exits are nearly the same (8 in direct lift vs. 9 in composite mode), a single lift loss curve is representative of both modes. The data appearing in Figure 2-1 is based on experimental test data of an eight jet model (Reference 2-1) and a ten jet NASA model (Reference 2-2). Adjustments applied to these data included those for increased pressure ratio ($\frac{P_T}{p}$ = 1.4 to 2.2), increased planform to jet exit area ratio ($\frac{S}{A_J}$ = 50 to 65), and a correction for low-to mid-wing position. The lift loss at an $\frac{h}{D_E}$ = 10 is 4% of maximum thrust. At landing gear heights of h/D_E = 1.19 and 1.26, Figure 2-1 shows lift losses of 15% and 15.5% for direct lift and composite modes respectively. The variation of the moment parameter $\frac{\Delta M}{T\bar{c}}$ with h/D_E shown in Figure 2-1 was obtained from Reference 2-5 which provided moment characteristics of a multiple jet configuration. From Figure 2-1 it may be noted that near ground ($h/D_E < 4.0$), the moment increases positively due to larger negative interference forces aft of the c.g. At landing gear height, $\Delta M/T\bar{c}$ reaches a maximum value of + 0.05.

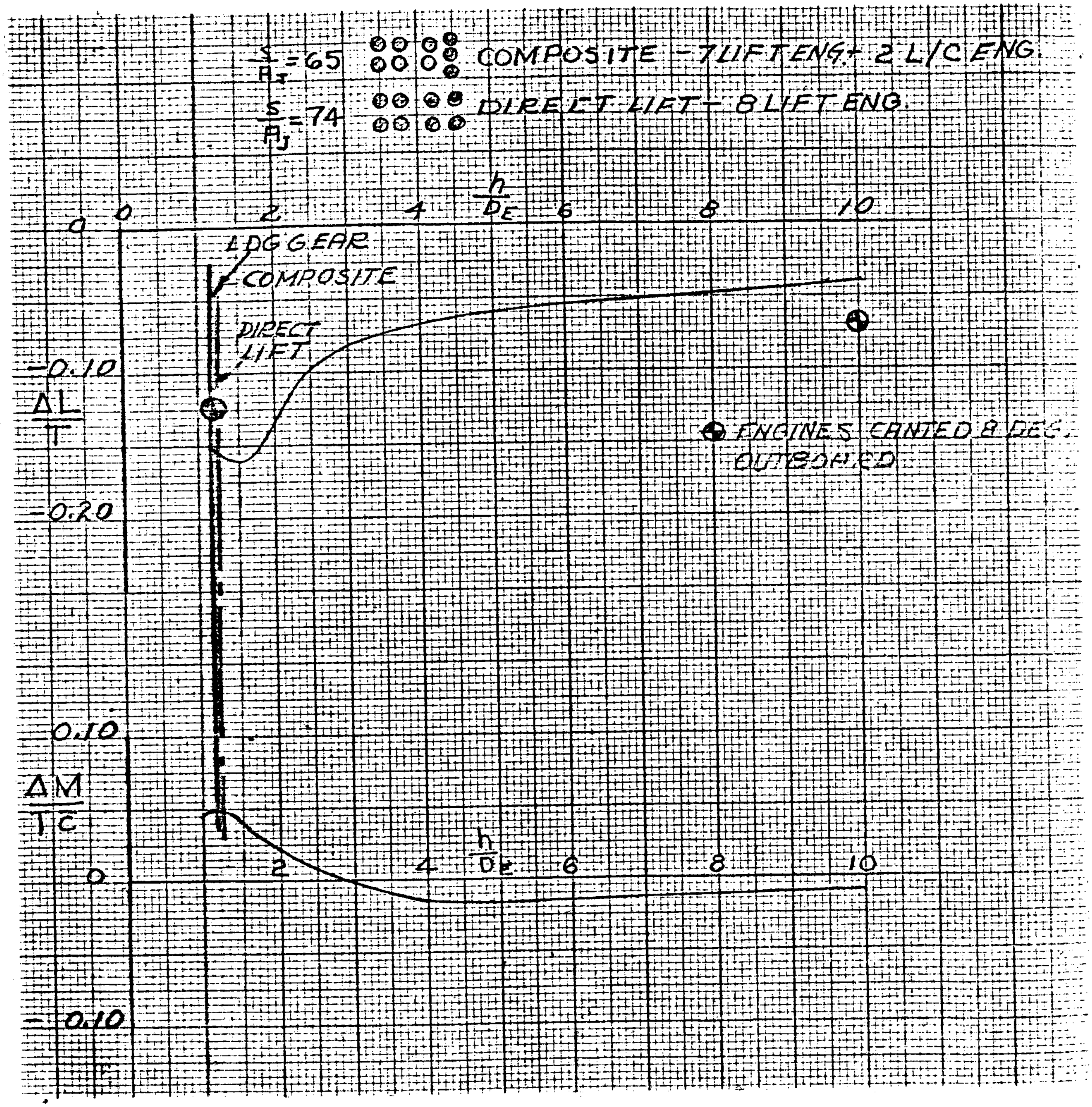


FIGURE 2-1. JET INDUCED INTERFERENCE LIFT & MOMENT IN HOVER - N-309

The interference lift loss curve in Figure 2-2 is representative of both direct lift and composite modes for the MOD T-39A configuration and is based primarily on the lift loss characteristics of a NASA ten jet test model (Reference 2-2) which has an $S/A_J = 50$ and $P_T/P = 1.4$. Similar corrections for S/A_J (50 to 74) and pressure ratio $\frac{P_T}{P}$ (1.4 to 2.2) were applied which resulted in a 6% lift loss at $h/D_E = 10$ and 16.5% at landing gear height.

One method which has been effective in reducing lift loss near the ground at approximately landing gear height is the outboard canting of the jet exit nozzles. Test results on a NASA 10 jet model (Reference 2-2) indicated that canting the jet exits outboard 10^o reduced maximum lift loss at $h/D_E = 1.5$ by 7.0%. This reduction includes the effect of a 1.5% loss in vertical thrust due to canting the thrust vector. However for $h/D_E > 4.0$, there was an increase in lift loss of 4.5%. The results of the NASA tests were applied to the N-309 and Mod T39A configurations and an 8 degree outboard cant of thrust was assumed which resulted in a 1% vertical thrust loss. The following lift loss characteristics shown by the symbols are presented in Figures 2-1 and 2-2. For the N-309 configuration in Figure 2-1 the lift loss $\Delta L/T$ amounts to 12.5% at landing gear height and 7% at $h/D_E = 10$. For the Mod T-39A configuration in Figure 2-2, the corresponding lift losses are $\Delta L/T = 11.5\%$ at landing gear height and 9% at $h/D_E = 10$.

The moment variation $\Delta M/T_{\bar{c}}$ with h/D_E in Figure 2-2 for the MOD T-39A is identical to that for the N-309 configuration. At a landing gear height $h/D_E = 1.5$, the value of $\Delta M/T_{\bar{c}} = 0.04$.

A multiple lift jet aircraft in transition experiences much larger interference lift loss forces than in hover. At forward speeds the jet exhaust plume is deflected rearward which causes a larger undersurface area to be affected by negative pressures. It has been suggested from flow visualizations (Reference 2-6) that the trailing jet plume consists of two contra-rotating vortices in the shape of a horseshoe which are responsible for the development of the negative pressures. Thus the magnitude of lift loss associated with a particular aircraft configuration would be highly dependent upon the amount of undersurface area rearward of the jet exits. Other effects which influence lift loss are the magnitude of jet velocity, jet deflection angle and tail downwash. The variation of lift loss $\Delta L/T$ with $\sqrt{\frac{\rho_\infty V_\infty^2}{\rho_j V_J^2}}$ shown in Figure 2-3 is representative of both direct lift and composite modes of the N-309 configuration. Since the N-309 planform is similar to the low swept wing configuration in Reference

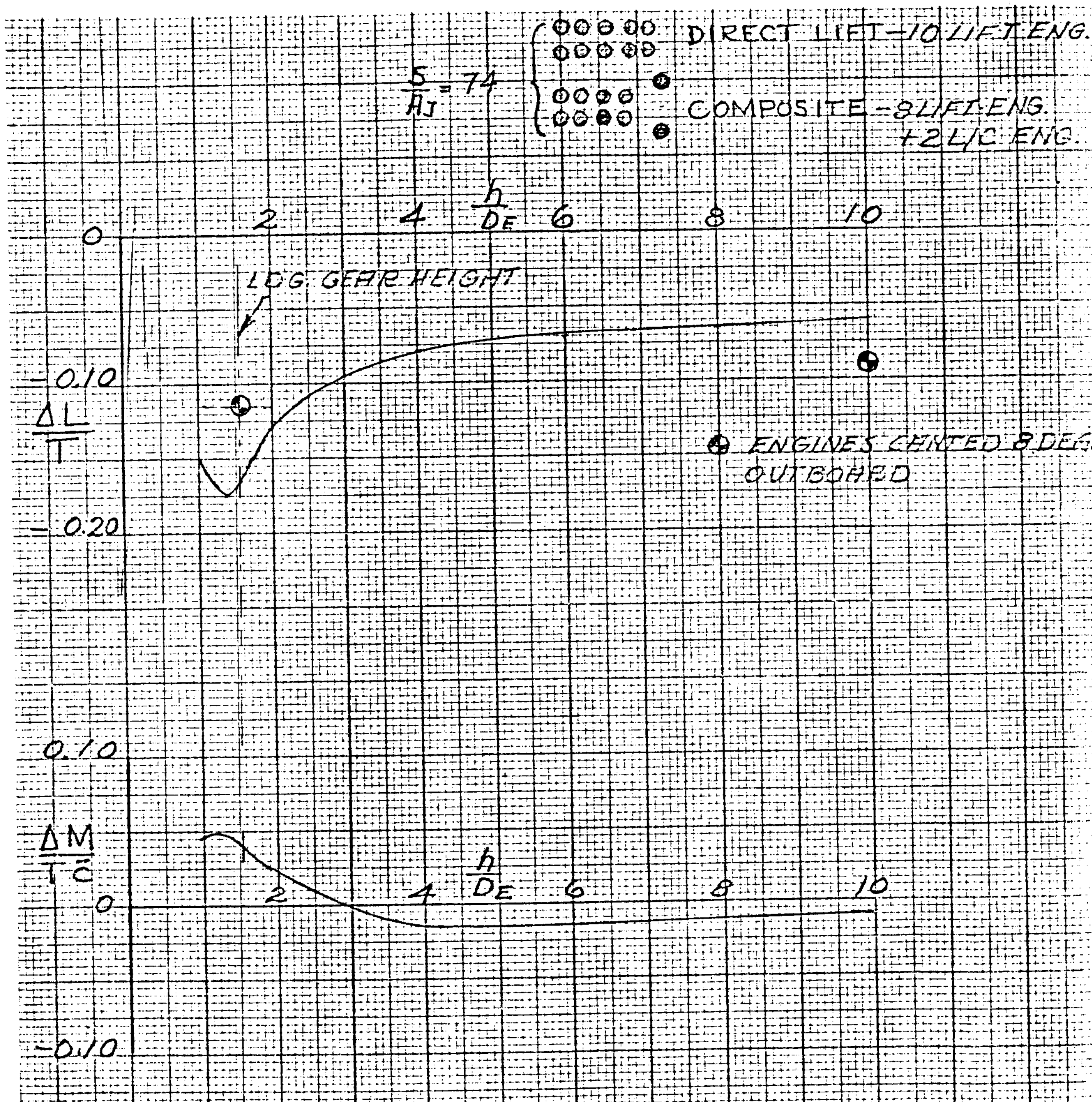


FIGURE 2-2. JET INDUCED INTERFERENCE LIFT & MOMENT IN HOVER - MOD. T-39A

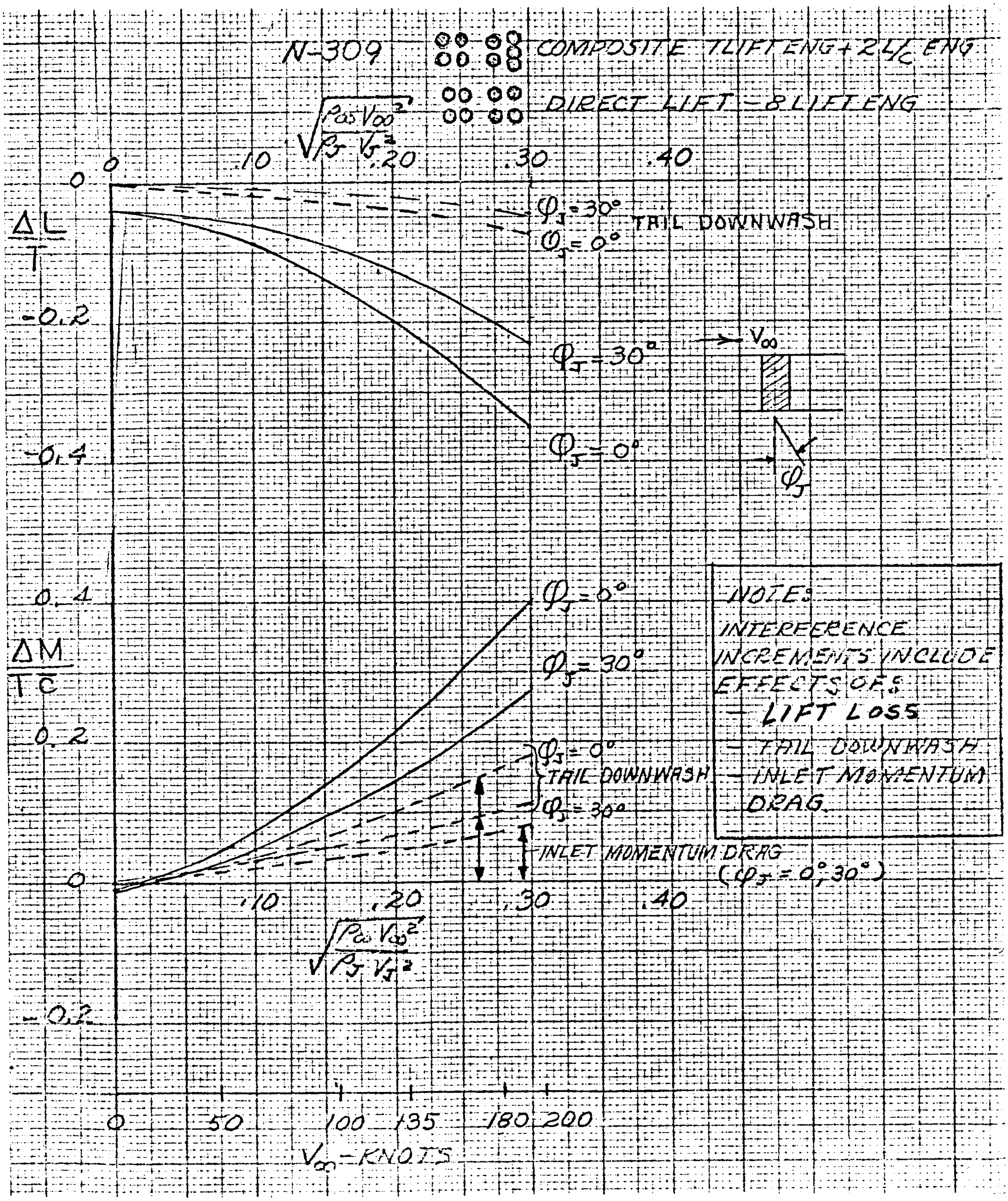


FIGURE 2-3. JET INDUCED INTERFERENCE LIFT & MOMENT IN TRANSITION - N-309

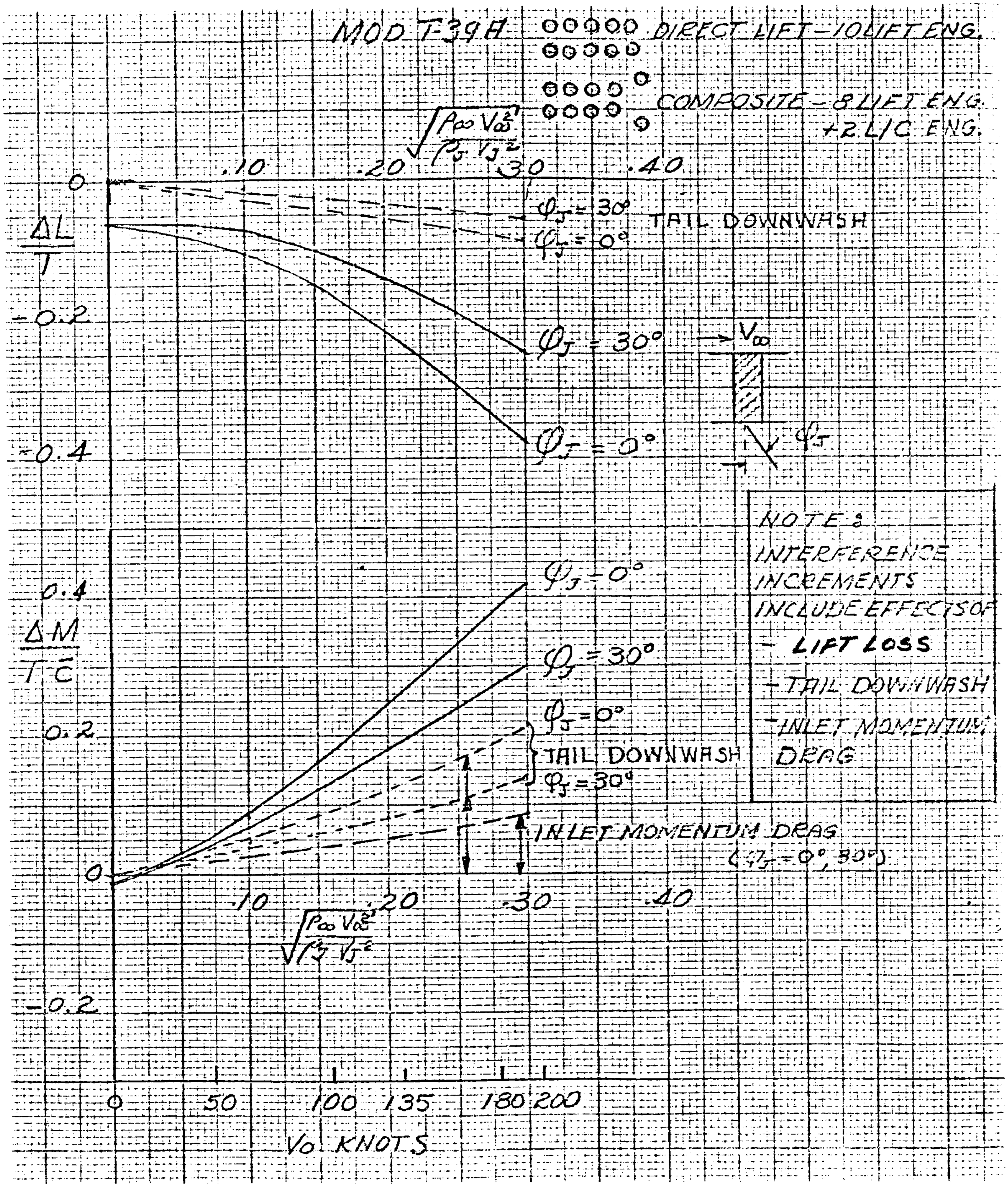


FIGURE 2-4. JET INDUCED INTERFERENCE LIFT & MOMENT IN TRANSITION - MOD. T-39A

2-7, lift loss data from this reference were used as a basis to establish lift loss characteristics. Effects of jet vectoring ($\varphi_J = 30^o$) and tail downwash (tail undeflected for trim) are also presented. Jet vectoring and tail downwash contributions were obtained from References 2-6 and 2-8. The incremental moment parameter $\Delta M/T_{\bar{c}}$ is shown in the lower half of Figure 2-3. The moment variation includes contributions of suckdown from Reference 2-7 inlet momentum drag, tail downwash (tail undeflected for trim) and jet vectoring ($\varphi_J = 30^o$). Similar interference lift and moment characteristics for the T-39A configuration are presented in Figure 2-4. Analyses of incremental lift and moment variations with jet vectoring from References 2-6 and 2-8 have indicated that interference lift and moment effects may be represented by the following empirical equations over the transition speed range.

$$\frac{\Delta L}{T} = .75\left(\frac{\Delta L}{T}\ \varphi_J = 0\right) \text{Cos}\ \varphi_J$$

$$\frac{\Delta M}{T_{\bar{c}}} = .75\left(\frac{\Delta M}{T_{\bar{c}}}\ \varphi_J = 0\right) \text{Cos}\ \varphi_J$$

Where: φ_J is the jet vector angle measured from vertical and factor (.75) applies at $\varphi_J > 0$.

2.1.2 Hot Gas Ingestion

A study of recent full scale hot gas ingestion tests as reported in Reference 2-3 has indicated that the phenomena of hot gas ingestion relative to multiple lift jet V/STOL aircraft is highly time- and configuration-dependent. The tests revealed that engine stall is not necessarily associated only with ingestion-prone configurations, since several stalls occurred in configurations which for the most part exhibited little or moderate ingestion tendencies. In most cases some thrust loss due to ingestion of hot gases was noted. For an analysis of test results in Reference 2-3, six multiple jet configurations with both delta and swept planforms showed ingestion thrust losses which varied from zero to approximately 10% of maximum thrust. The trend of the data showed that the higher thrust losses occurred at $\frac{h}{D_E} \cong 2.0$. Perhaps of primary significance is that there now is a better understanding of the overall problem of hot gas ingestion and, as a result, means for alleviating or eliminating hot gas ingestion is at hand.

Of major importance in minimizing hot gas ingestion effects is the shielding of the inlets from the entrainment of hot gases. This may be accomplished by locating the wing as low to the plane of the jet exits as possible, although consideration must also be given to lift losses. Wing planforms of large root chord are preferable to act as a shield. Another essential is the close grouping of the jet exits so that the exhaust plumes coalesce beneath the wing surfaces to act as a single jet. In this way the jet upon impinging with the ground would form a low outward flowing sheet. Jets that are spaced far apart (as is the case in the MOD T-39A lift-cruise engine arrangement) may cause a jet "fountain" to be developed. The "fountain effect" is an upward flow of hot gas which, if unshielded, could be entrained by the nearest inlet. In a landing transition, the most forward lift jet engines may be most susceptible to ingestion since exhaust gases would precede the aircraft in the landing approach. One method to alleviate hot gas ingestion that has been investigated in Reference 2-9 is the aerodynamic shield. The aerodynamic shield is a fine sheet of air issuing from the side of the fuselage below the inlet to prevent hot gas ingestion.

It is concluded that extrapolation of hot gas ingestion data from Reference 2-3 to new configurations such as the N-309 and MOD T-39A would only indicate gross tendencies. No changes in lift loss or moment change from hot gas ingestion are included in the data of Figures 2-1 and 2-2. However, the two new aircraft have been configured such as to minimize hot gas ingestion and to prevent compressor stall.

2.2 CONVENTIONAL FLIGHT AERODYNAMICS

2.2.1 Basic Wing Lift

The N-309 wing was selected primarily for good low speed aerodynamic characteristics, since high speed is not a requirement. Significant items are high maximum lift for low stall speeds and low induced drag for safe single engine flight. A 12-to 13-percent thick wing loading of 85 psf, an aspect ratio of 6, and 0.4 taper ratio best suits these requirements. In addition, the quarter chord is swept 20^{o} to provide a reasonably high drag divergence Mach number. The T-39A wing has not been modified, because it meets flaps up stall speed requirements. However, there is only a 30 knot difference between clean and landing stall speeds.

2.2.2 High Lift System

Leading and trailing edge high lift devices were employed on the N-309 to provide the NASA desired 40 knot stall speed spread between the clean and landing configurations. A full span 20% chord leading edge flap is combined with a 73% span, 25% chord trailing edge single slotted trailing edge flap with a drooped shroud.

Norair tests of a drooped shroud on a single slotted flap showed it to be effective as a Fowler or double slotted flap for high lift.

The T-39A wing high lift system comprises a leading edge full span, constant chord leading edge and a 60% span 28% chord trailing edge single slotted flap.

Lift curves for both airplanes are presented in Figures 2-5 through 2-8. N-309 lift was estimated by methods of References 2-10 and 2-11. T-39A maximum lift was obtained from the pilot's flight manual (Reference 2-12 and $C_{L_{\tilde{\alpha}}}$ was estimated from Reference 2-10. These curves are not trimmed, since trimmed lift increments will be positive or too small to significantly change lift, and the center of gravity shift with weight is also small.

2.2.3 Drag

2.2.3.1 MINIMUM DRAG. Airplane minimum drag is composed of a buildup of individual component skin frictions plus profile drags. They are combined with interference drag and compressibility drag increments at high Mach numbers to make up the clean airplane minimum drag. The skin friction drag level is based on a 1.60×10^6 Reynolds number per foot. (150 knots, M ≈ .23.)

Total airplane minimum drag for the N-309 and T-39A is shown in Figures 2-9 and 2-10 respectively. Drag estimates are based on data from References 2-10 and 2-13. The coefficients are defined and added for total minimum drag as shown below.

Drag Coefficient

$$\Delta C_D \equiv C_f \left[\frac{C_f}{C_{f_{F.P.}}} \right] \frac{S_{WET}}{S}$$

Minimum Drag

$$C_{D_{MIN}} = \Delta C_{D_{WING}} + \Delta C_{D_{FUS}} + \Delta C_{D_{HOR}} + \Delta C_{D_{VERT}} + \Delta C_{D_{INT_{W-B}}}$$

$$\Delta C_{D_{INT_{H-V}}} + C_{D_{COMP.}}$$

Minimum drag was assumed constant until compressibility effects were present. Then the individual compressibility effects of each of the aircraft components was added to the minimum drag to construct the drag rise.

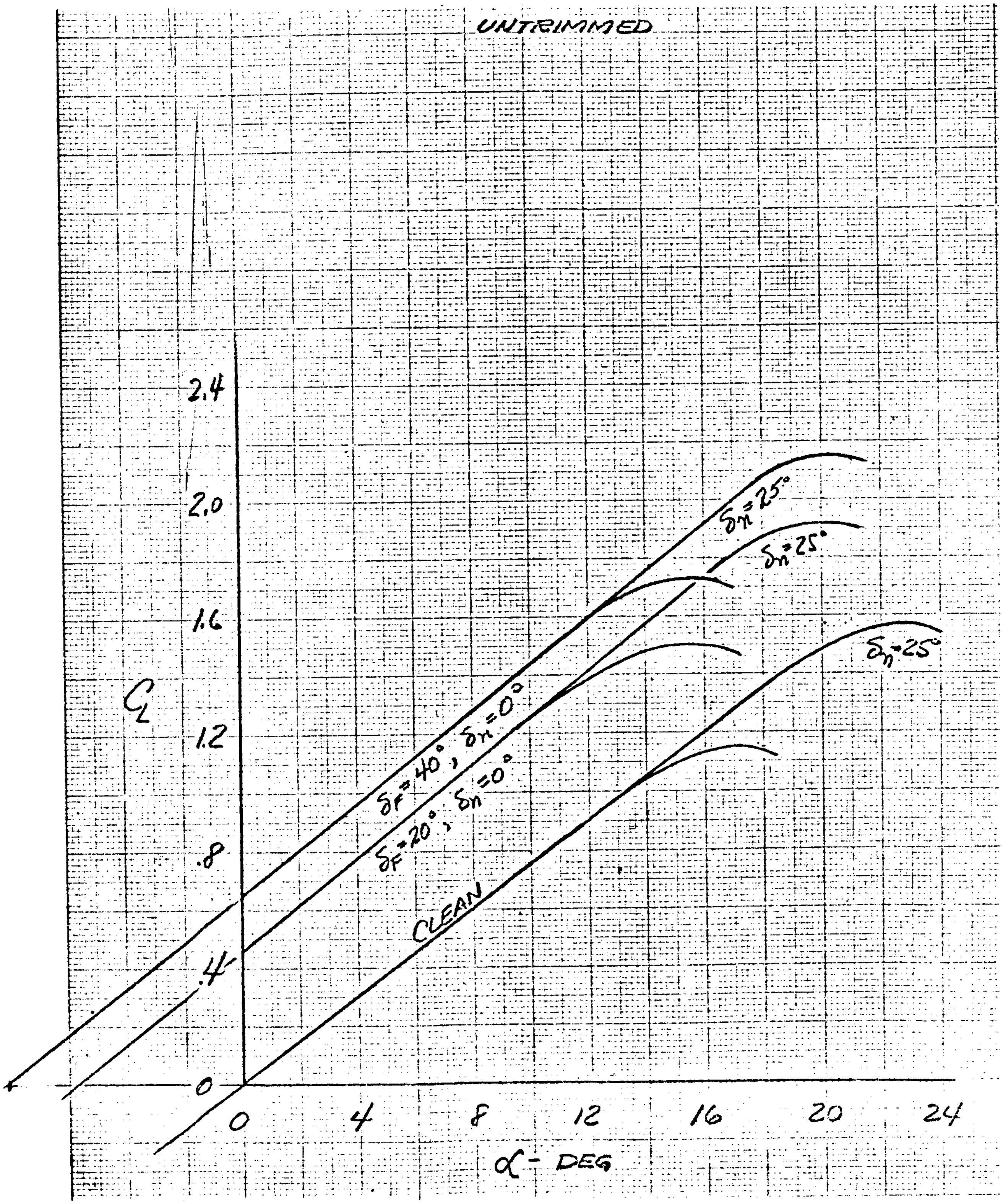


FIGURE 2-5. N-309 LIFT CURVES

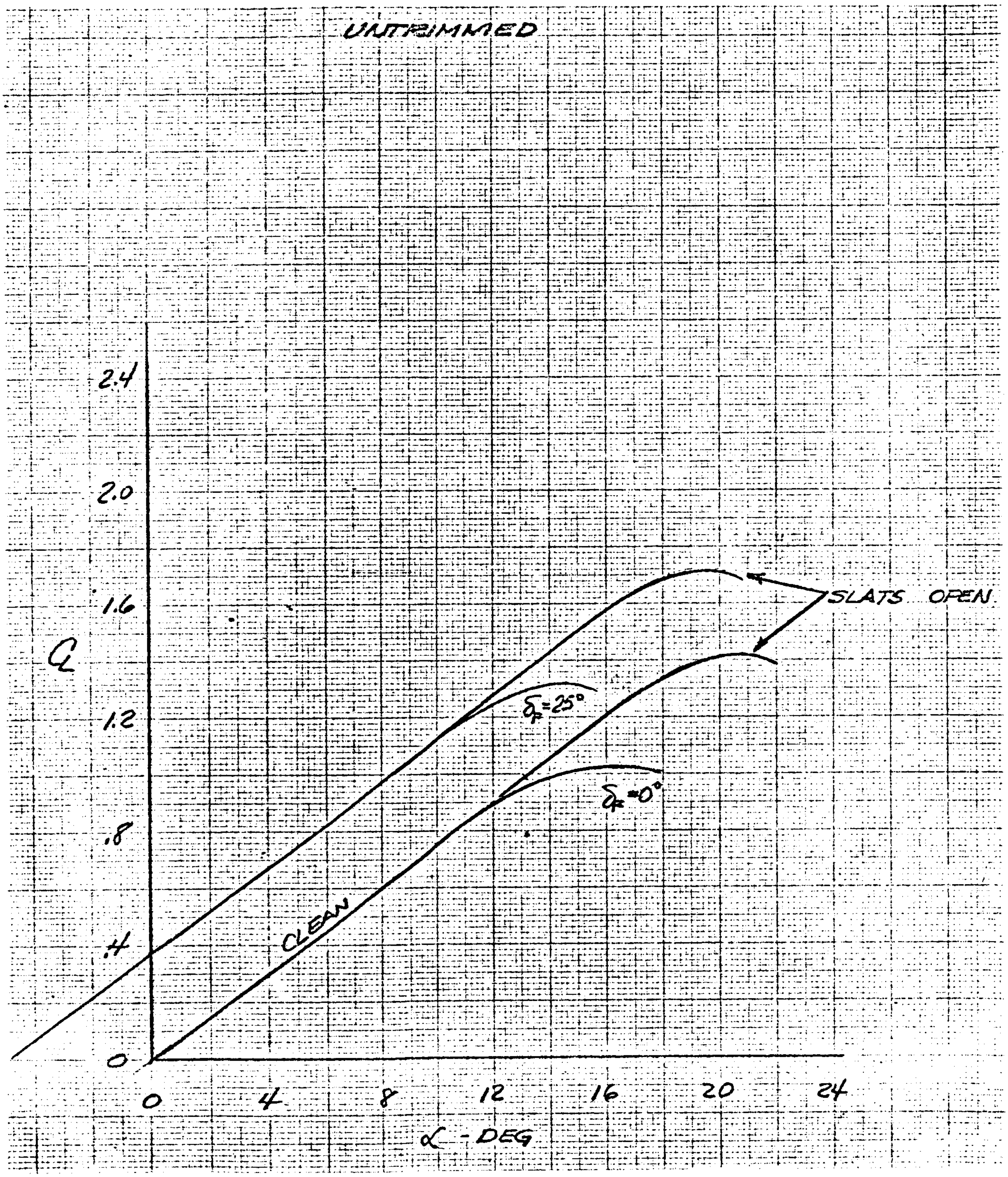


FIGURE 2-6. T-39A (MOD.) LIFT CURVES

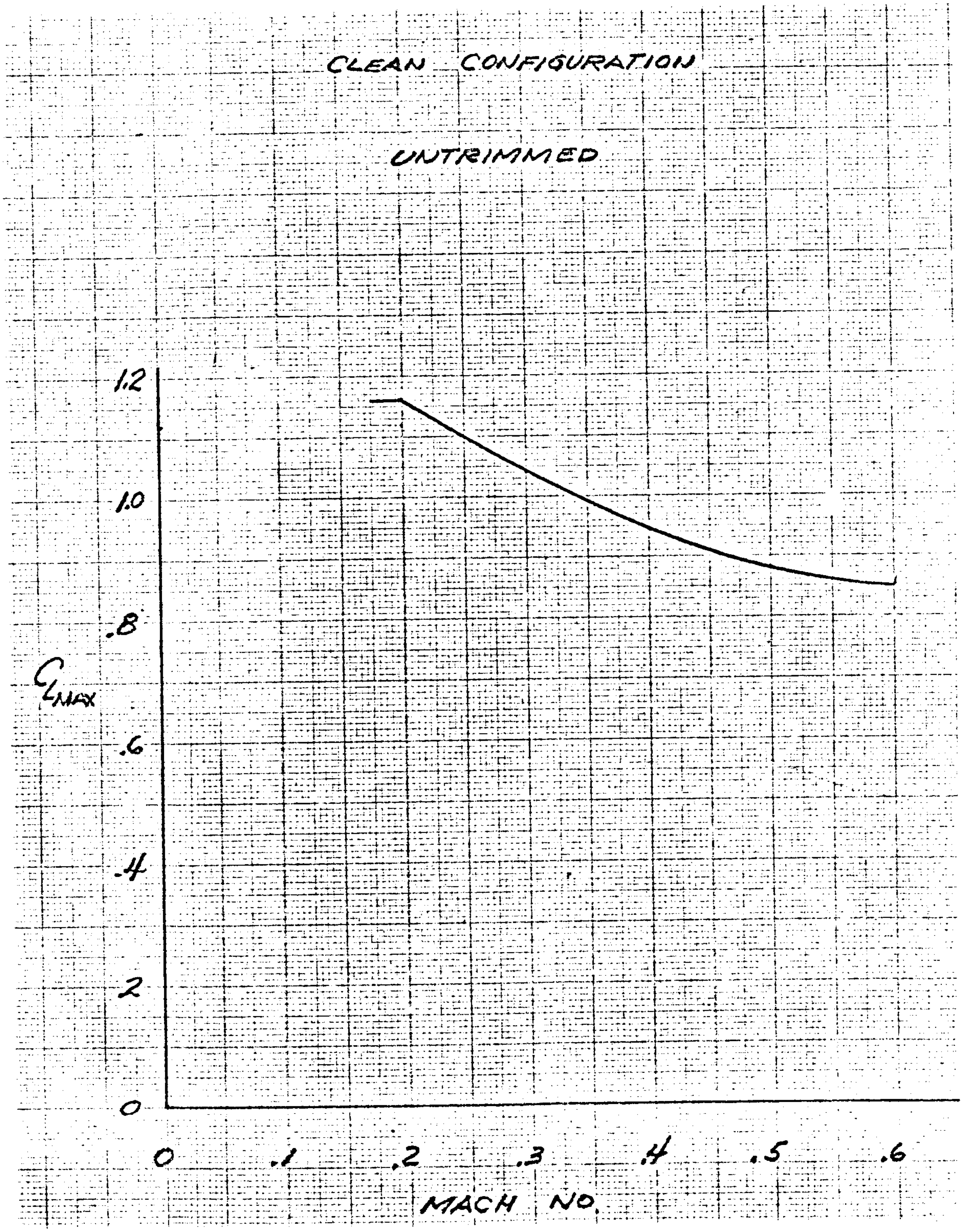


FIGURE 2-7. N-309 $C_{L_{MAX}}$ VS MACH NO.

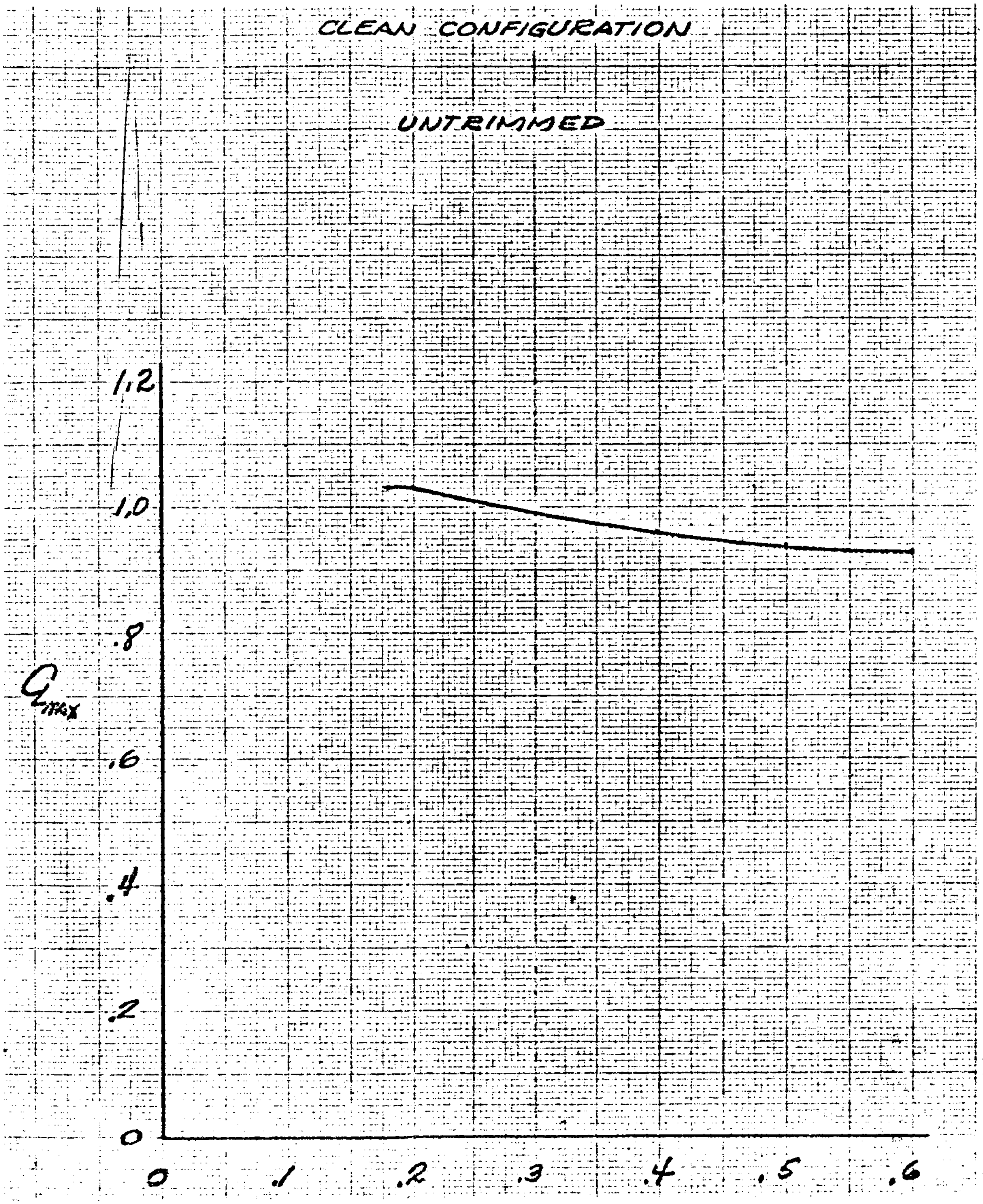


FIGURE 2-8. T-39A (MOD.) $C_{L_{MAX}}$ VS MACH NUMBER

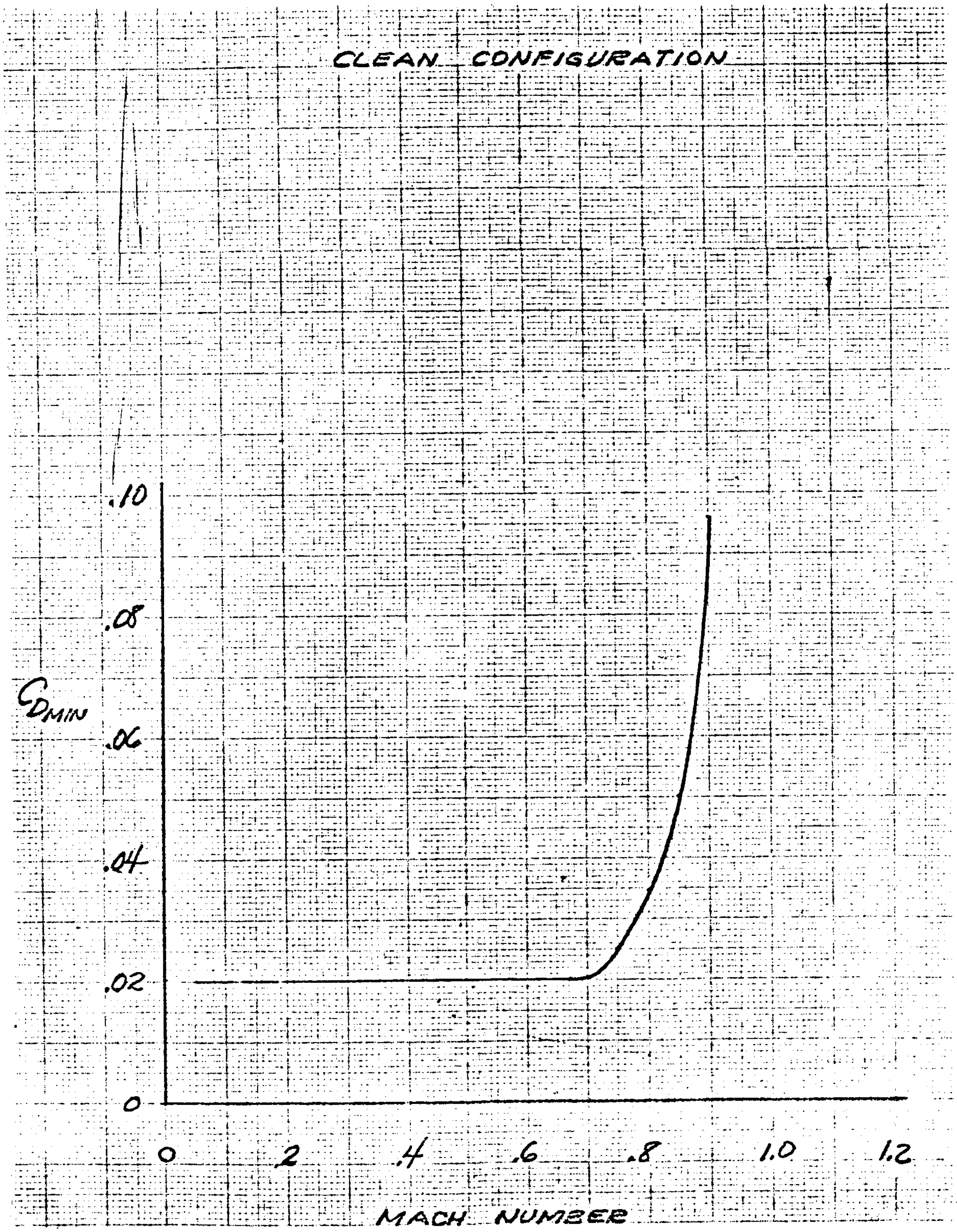


FIGURE 2-9. N-309 MINIMUM DRAG

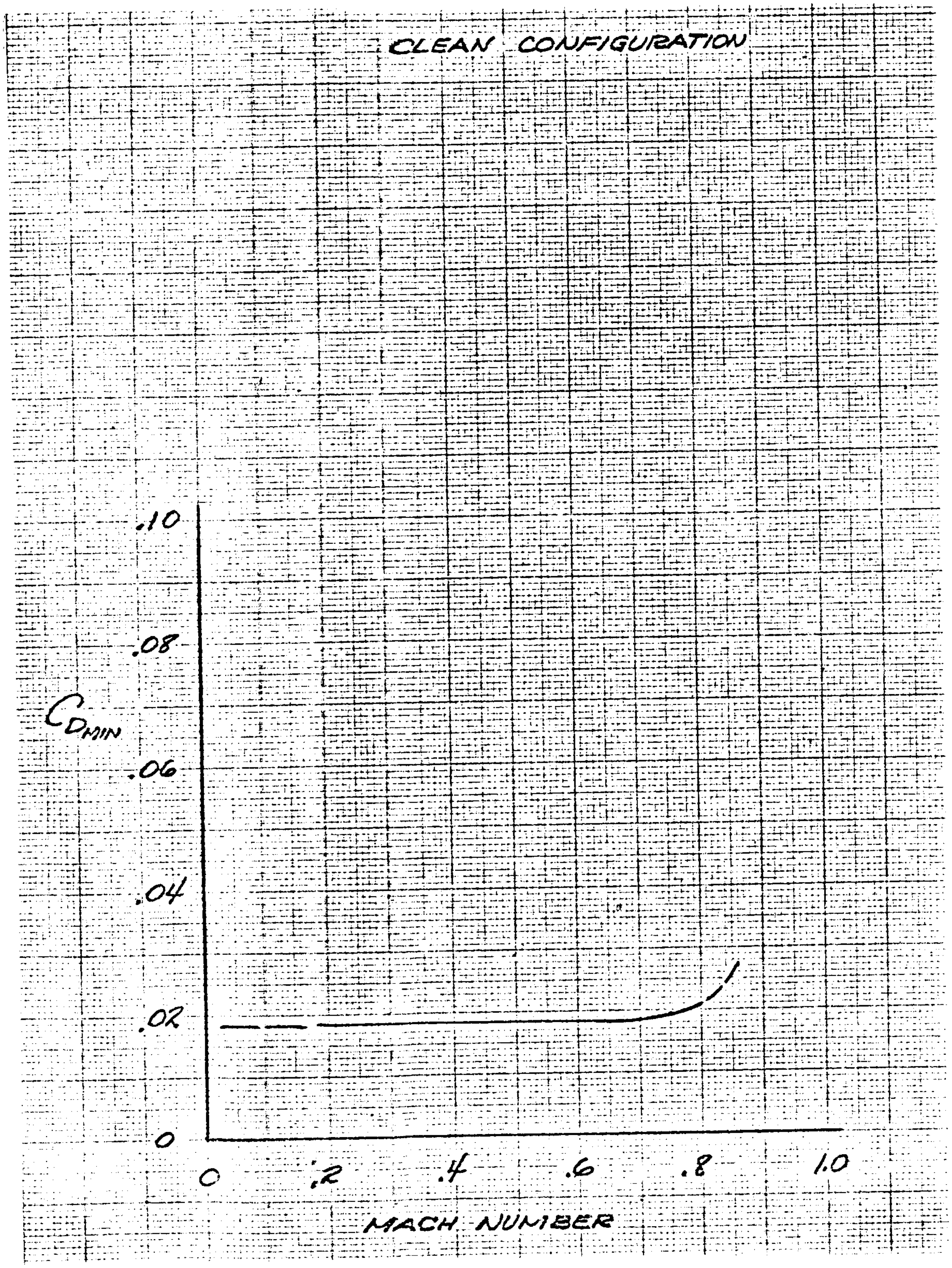


FIGURE 2-10. T-39A (MOD.) MINIMUM DRAG

2.2.3.2 Drag Polars. Drag polars are constructed for the clean, approach and landing configurations for both aircraft. For the T-39 approach and landing, slat and flap positions are the same (slats open and 100% flaps). Drag polars are not trimmed, but trim drag will not significantly change airplane performance.

The clean drag polar is represented by the following equation. Approach and landing

$$C_D = C_{D_{MIN}} + \frac{{C_L}^2}{\pi ARe}$$

configuration polars were constructed by adding landing gear and flap parasite drag increments to the clean polar as shown below:

$$C_D = C_{D_{MIN}} + \Delta C_{D_{gear}} + \Delta C_{D_{flaps}} + \frac{{C_L}^2}{\pi ARe}$$

Wing leading edge flap or slat drag was assumed negligible for these estimates.

Inasmuch as the N-309 and the T-39A have similar wing geometries, airplane efficiencies are alike. For the clean configurations, below the drag break, airplane efficiency is 0.82 for the N-309 and 0.77 for the T-39A. N-309 drag polars for the clean, approach and landing configureations are on Figures 2-11 and 2-12 and T-39A polars are on Figures 2-13 and 2-14.

Minimum drag was established by methods in Reference 2-10 and drag due to lift, landing gear and flap drags were estimated from References 2-11, 2-13, and 2-14.

2.3 PERFORMANCE

2.3.1 Hover Endurance

Hover endurance for the N-309 and T-39A for composite and direct lift operation is presented in Table 2-1. These data are based on out-of-ground effect thrust levels plus an interference lift loss margins, a 5% service tolerance on engine SFC, and the latest guaranteed minimum engine performance. Ninety percent of the reaction control thrust is included as lift. NASA criteria complied with include 3.75 g design load factor for the new aircraft, 1.25 times the resulting modified aircraft load factor for new portions of the aircraft, sea level, 80°F ambient conditions and weight contingencies.

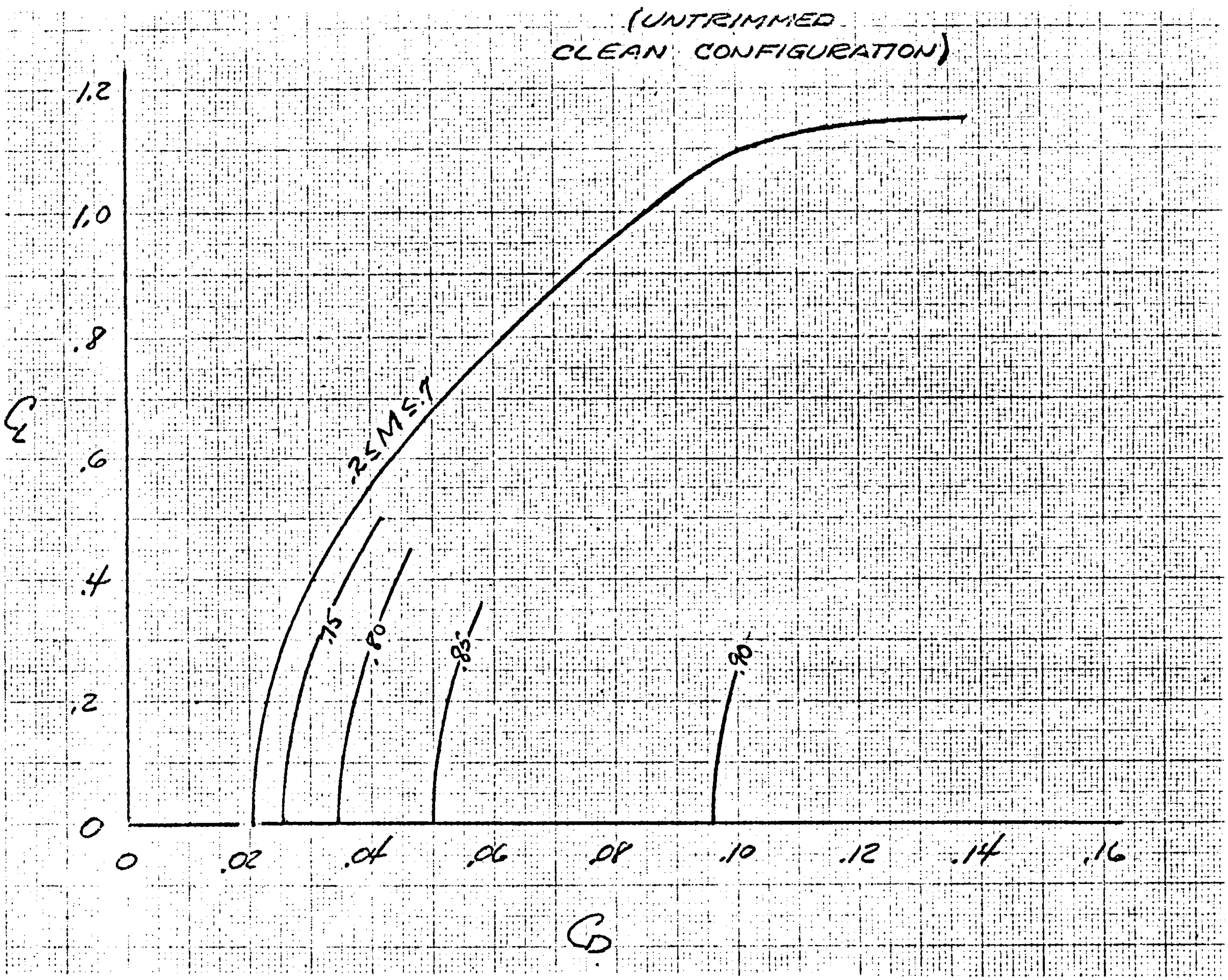


FIGURE 2-11. N-309 DRAG POLARS

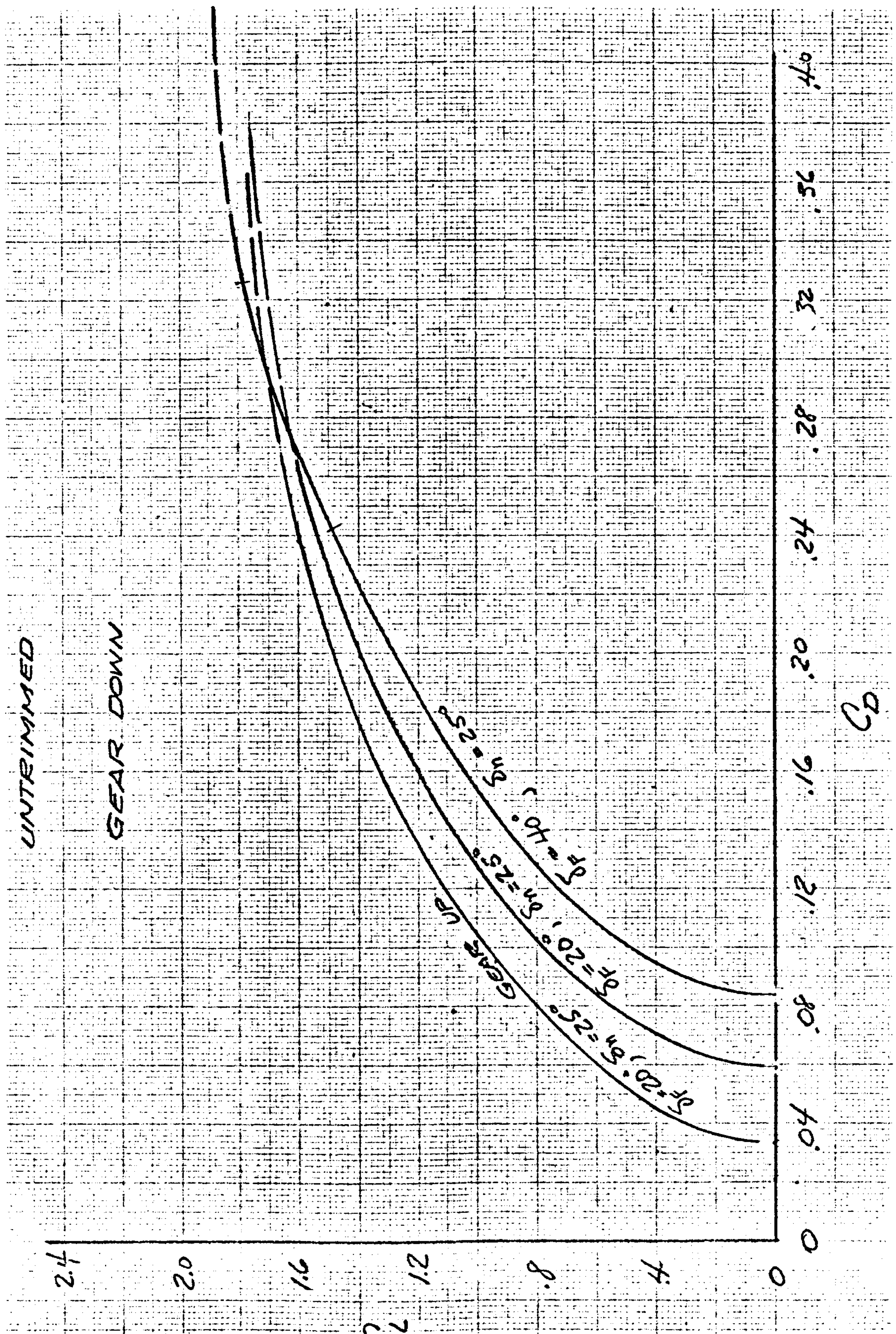


FIGURE 2-12. N-309 LOW SPEED DRAG POLARS

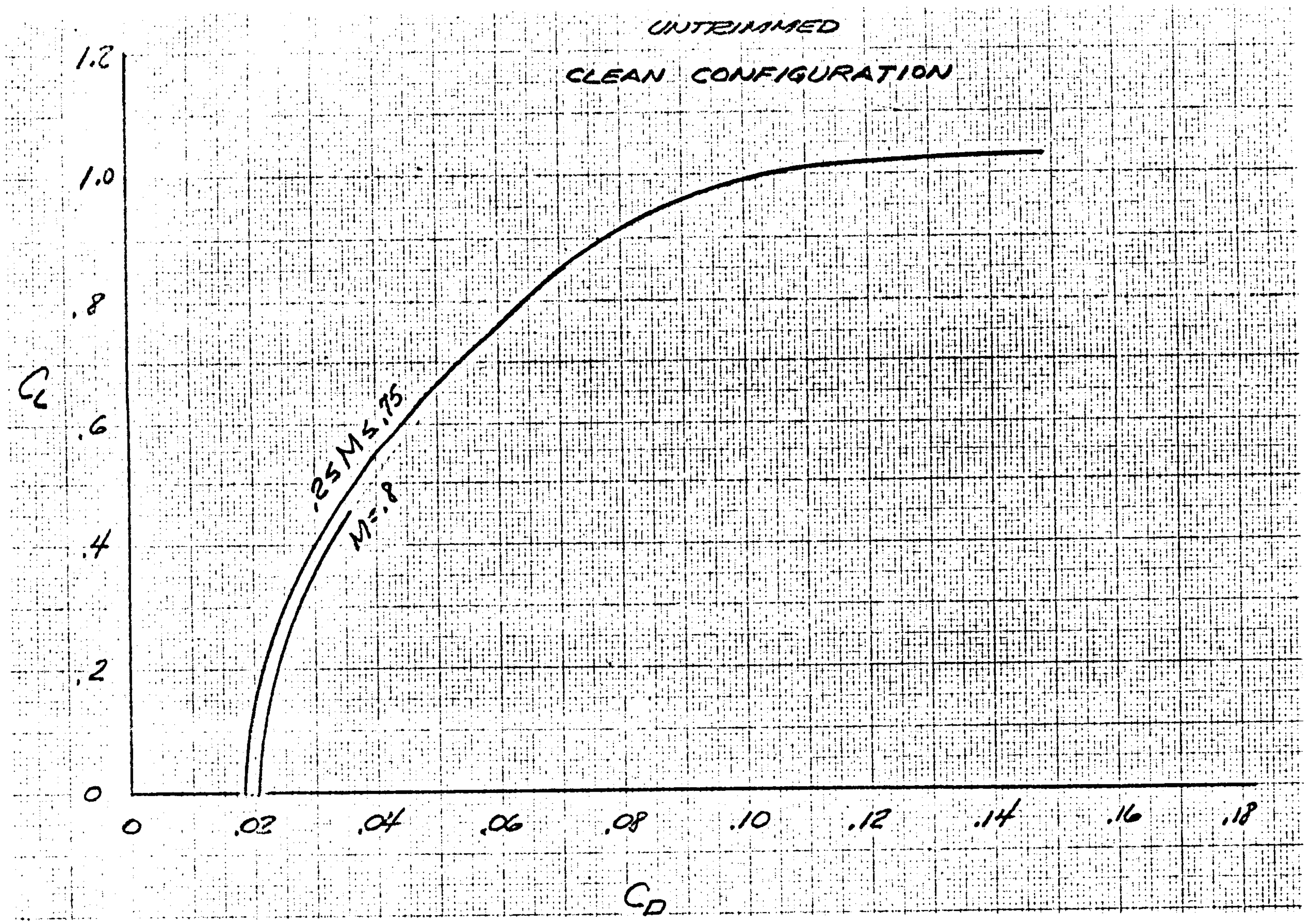


FIGURE 2-13. T-39A (MOD.) DRAG POLARS

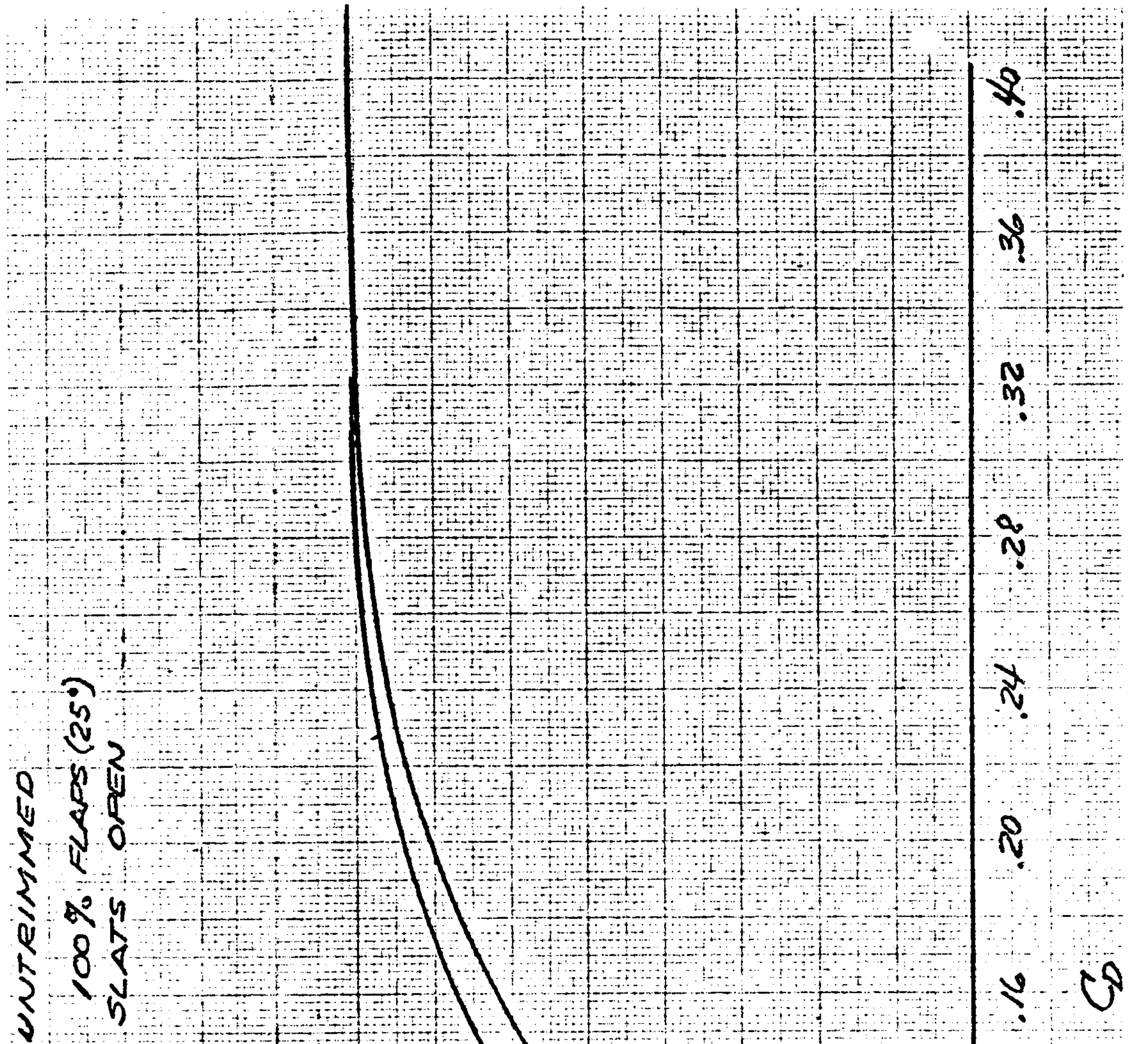


FIGURE 2-14. T-39A (MOD.) LOW SPEED DRAG POLARS

TABLE 2-1. HOVER ENDURANCE

SEA LEVEL, 80°F DAY

NASA THRUST MARGINS PLUS
INTERFERENCE LIFT LOSS

Designation Engine Configuration	N-309 8/7+2 (J85-19)	Mod. T-39A 10/8+2 (J85-19)
Composite Time (Min.)	13.4 (W = 18,000 lb)	12.7 (W = 20,200 lb)
Direct Lift Time (Min.)	8.0 (W = 16,300 lb)	10.8 (W = 20,200 lb)

NOTE: Weight contingency included.

N-309 direct lift hover time is 5.4 minutes less than composite time because the latter with one more lift engine carries more fuel, and cruise engine idle fuel (600 lb/hr) had to be accounted for in the direct lift mode. Hover times for the T-39A differ by 1.9 minutes, because added lift engines for the direct lift configuration requires off loading fuel. In the direct lift configuration both aircraft recover 300 lbs. of fuel that replaces additional research payload.

Improvements in the propulsion system installation have analytically resulted in a thrust gain, subsequent to the airplane's final weight statement. If later experiments show the thrust increases, it is estimated that the allowable hover weight could be raised by 2% with an associated 8% or better greater hover time.

2.3.2 Hover T/W Margins

NASA required and aircraft available thrust to weight margins, with originally estimated weights and total thrust, at specified hover control conditions are compared on Table 2-2. Norair estimated lift losses are included with NASA free air and ground effect thrust margins. N-309 and T-39A thrust available either meet or exceed the NASA specified thrust margins for all control conditions.

TABLE 2-2. HOVER T/W MARGINS

ENGINES OPERATING	CONTROL % MAX			N-309				MOD. T-39A			
				REQUIRED *		AVAILABLE		REQURED *		AVAILABLE	
	PITCH	ROLL	YAW	IGE	OGE	IGE	OGE	IGE	OGE	IGE	OGE
ONE OUT	20	50	20	-	1.09	-	1.10	-	1.11	-	1.11
ALL	80		50	1.20	-	1.24	-	1.21	-	1.23	-
ALL	80			-	1.09	-	1.24	-	1.11	-	1.23
ALL	50			1.20	-	1.24	-	1.21	-	1.23	-
ALL	50			-	1.19	-	1.24	-	1.21	-	1.23

* INTERFERENCE LIFT LOSS INCLUDED IN NASA T/W MARGIN REQUIREMENTS

2.3.3 Transition Acceleration

The acceleration performance comparisons presented in Figure 2-15 were developed according to the initial conditions of maximum take off weight (18,000 lbs. for N-309 and 20,200 lbs. for the MOD T-39A) maximum thrust at sea level and 80°F, and constant flight path angle $\gamma = 0^{o}$. A computer program provided acceleration characteristics which included effects of conventional aircraft lift and drag as well as interference lift and inlet momentum drag. Thrust components due to angle of attack variation were accounted for. For small angle of attack, the equations of motion along the flight path and normal to it are

(1) $$A_{\substack{FLIGHT\\PATH}} = \frac{g}{W}\left[T_H - T_V\,\alpha - D_{AIRPLANE} - D_{MOMENTUM} - W\,SIN\,\gamma\right]$$

and

(2) $$T_H\,\gamma + T_V + L_{AIRPLANE} - \Delta L - W\,COS\,\gamma = 0$$

where:

T_H and T_V are thrust components, parallel and normal to aircraft longitudinal axis.

ΔL - The interference lift loss

L_A, D_A - Conventional airplane lift and drag

$D_{MOMENTUM}$ - Inlet momentum drag of all operating engines

Figure 2-15 shows acceleration-velocity and velocity-time plots of four configurations of the N-309 configuration and one of the MOD T-39A configuration. Comparison of configurations 1 and 3 shows the advantage of flaps for improving acceleration and reducing time in transition. Initially, acceleration is 0.56 g's. At 200 knots, the 20° flap configuration shows an acceleration of 0.265 g's while the zero flap case shows 0.185 g's. The corresponding transition times are 26.5 and 31.5 seconds. This improvement is due to a larger forward thrust component made possible by negative angles of attack in transition for the same required lift.

Configuration 4, which consists of seven lift engines and two lift cruise engines with spherical nozzles all set at 28 degrees, is compared with configuration 1 which also has seven lift engines with spherical nozzles set at 28°, but the two lift cruise engine nozzles are canted 10 degrees aft with doors deflected aft 18 degrees. In

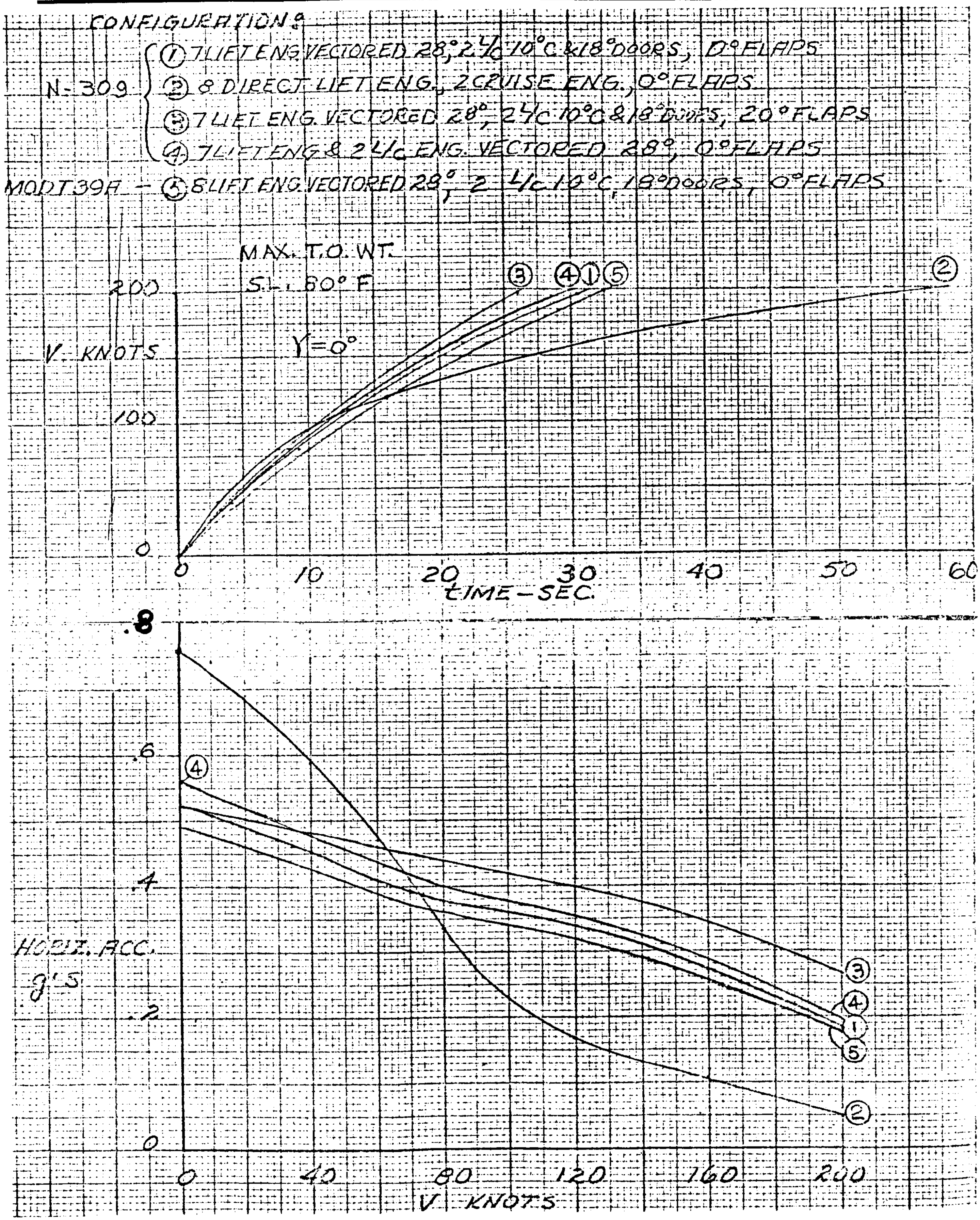


FIGURE 2-15. ACCELERATION TRANSITION PERFORMANCE

this comparison the acceleration of configuration 4 is approximately 7% higher than that of 1 and transition is accomplished in 30 seconds. Maximum initial acceleration of the direct lift engine, configuration 2, was achieved by pitching the nose down 21 degrees so that a forward thrust component could be obtained from the lift engines. This resulted in an initial acceleration of 0.76 g's. However, as angle of attack is is reduced at forward speed, acceleration drops off rapidly and net horizontal thrust diminishes. The time required to complete transition is 58 seconds. In comparing configuration 1, N-309, with 5, the MOD T-39A, it may be observed that the N-309 has slightly better acceleration characteristics throughout transition resulting in shorter transition time (31.5 seconds versus 32.5 seconds). Except for the direct lift configuration 2, the angle of attack variation for all other configurations ranged between $-3 \leq \alpha \leq 3$ degrees.

A comparison plot to show the effect of flight path angle on N-309 acceleration performance is presented in Figure 2-16. For a flight path angle of 10° the horizontal acceleration is 0.38 g's at zero forward velocity which reduces to 0.03 g's at 200 knots. For a flight path angle $\gamma = 0^{\circ}$ the corresponding acceleration performance is 0.525 g's at zero forward speed and 0.185 g's at 200 knots. Angle of attack over the speed range from zero to 200 knots varied from -2.9 degrees to + 1.8 degrees.

The acceleration performance shown is representative of the specific configuration arrangement chosen; i.e., flap angle and nozzle angle, engine cant and exit door deflection. Variation of these parameters within the limits set by efficient engine performance and feasible mechanical design are possible to improve transition performance. From the preceding analysis it is evident that flaps permit lift jet aircraft to be flown at higher acceleration which results in shorter transition time.

If additional acceleration capability is found necessary for minimum time during transition, the following approach could be considered. With some alteration in the design to accommodate an aft cant in the tailpipes of the lift engines, the spherical nozzles could give 15° forward vector for deceleration and 41° aft vector for acceleration since the total available travel in the nozzles is 56°. The resulting maximum acceleration, with half flaps at constant altitude, would be of the order of 0.6 g at hover decreasing to about 0.4 g at 200 knots. The time to 200 knots would then be approximately 20 seconds. If only 0.4 g were used throughout transition, the time to 200 knots would still be under 27 seconds and the time to 1.5 stall speed would be about 23 seconds.

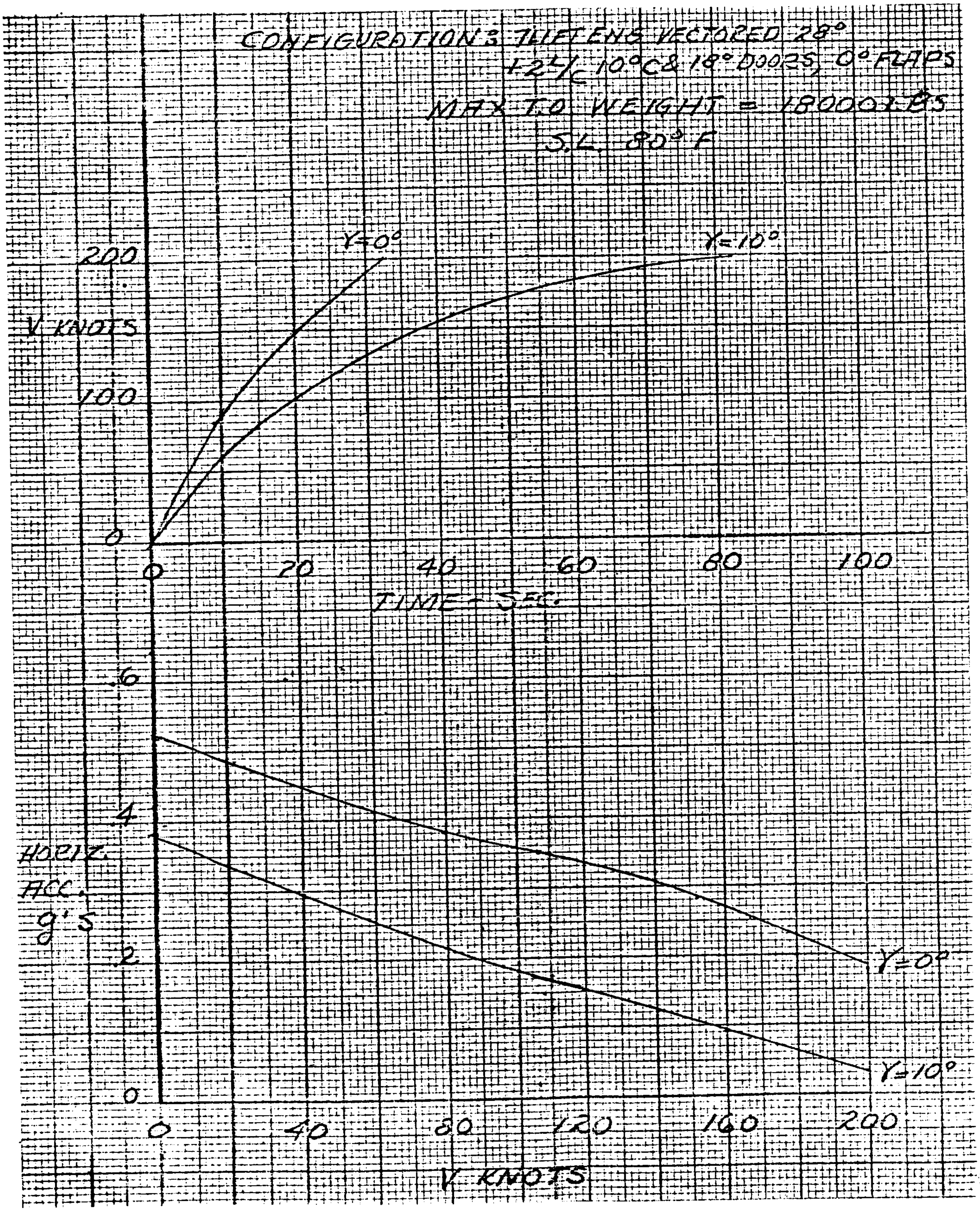


FIGURE 2-16. ACCELERATION PERFORMANCE COMPARISON – N-309

2.3.4 Transition Deceleration

The deceleration performance shown in Figures 2-17 and 2-18 for the N-309 and MOD T-39A configurations in composite mode is based on comparable initial configuration characteristics. The spherical nozzles of the lift engines (7 for the N-309 and 8 for the MOD T-39A) are vectored forward 15 degrees and the lift-cruise engine nozzles in lift mode remain canted 10 degrees aft. The performance was developed for a flight path angle $\gamma = 0$ degrees and two angles of attack: $\alpha = 0$ degrees and the angle of attack corresponding to 5 degrees below stall angle. This angle is 12° for the N-309 and 14° for the MOD T-39A configuration. The flaps are set 20° for N-309 and 25° for MOD T-39A and the landing gear is down. Lift engine thrust at the start of deceleration is a low percentage of maximum thrust and is varied in accordance with the lift requirements necessary to maintain constant altitude.

At a speed of 160 knots, the initial deceleration of the N-309 at a gross weight of 18,000 pounds is -0.35 g's at $\alpha = 0^{\circ}$ and the transition time is 30.5 seconds. At $\alpha = 14^{\circ}$ the initial deceleration at 100 knots is 0.28 g's which increases to a maximum of 0.46 g's at 40 knots. The time required in transition is 13.5 seconds. The deceleration performance developed according to the equations presented under acceleration includes the effects of interference lift loss, inlet momentum drag, thrust variation and thrust vector change with angle of attack.

Although provision is made for a thrust reverser, airplane deceleration with a reverser is not recommended. Deceleration can be controlled by airplane attitude, thrust level or combinations thereof. Moreover the foregoing deceleration times are low enough not to require additional decelerating devices. In addition to a weight penalty, the thrust reverser can direct hot exhaust gases over tail surfaces. This requires heat protection for these surfaces, and at very low speeds there can be significant tail aerodynamic forces. There is also the probability that deflected exhaust gases will reach the lift engine inlets and add to the hot gas ingestion problem.

2.3.5 Conversion

Thrust margins are necessary for all possible emergency situations during the conversion maneuver. The following cases have been investigated to determine vehicle capabilities during engine out emergencies.

N-309 thrust required and available is presented in Figure 2-19 at design gross weight, approach flaps, and unvectored lift engines during composite flight.

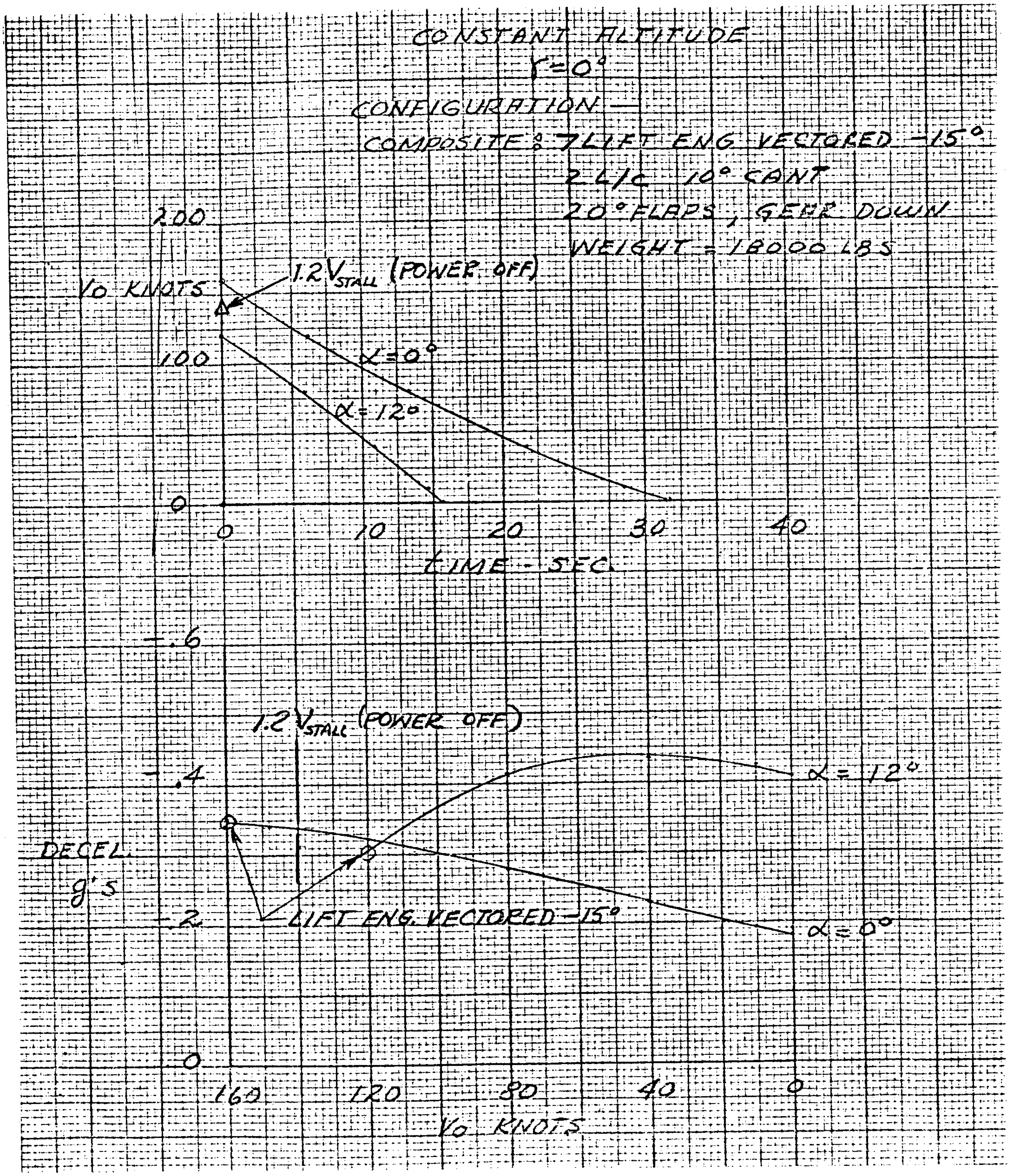


FIGURE 2-17. DECELERATION TRANSITION PERFORMANCE - N-309

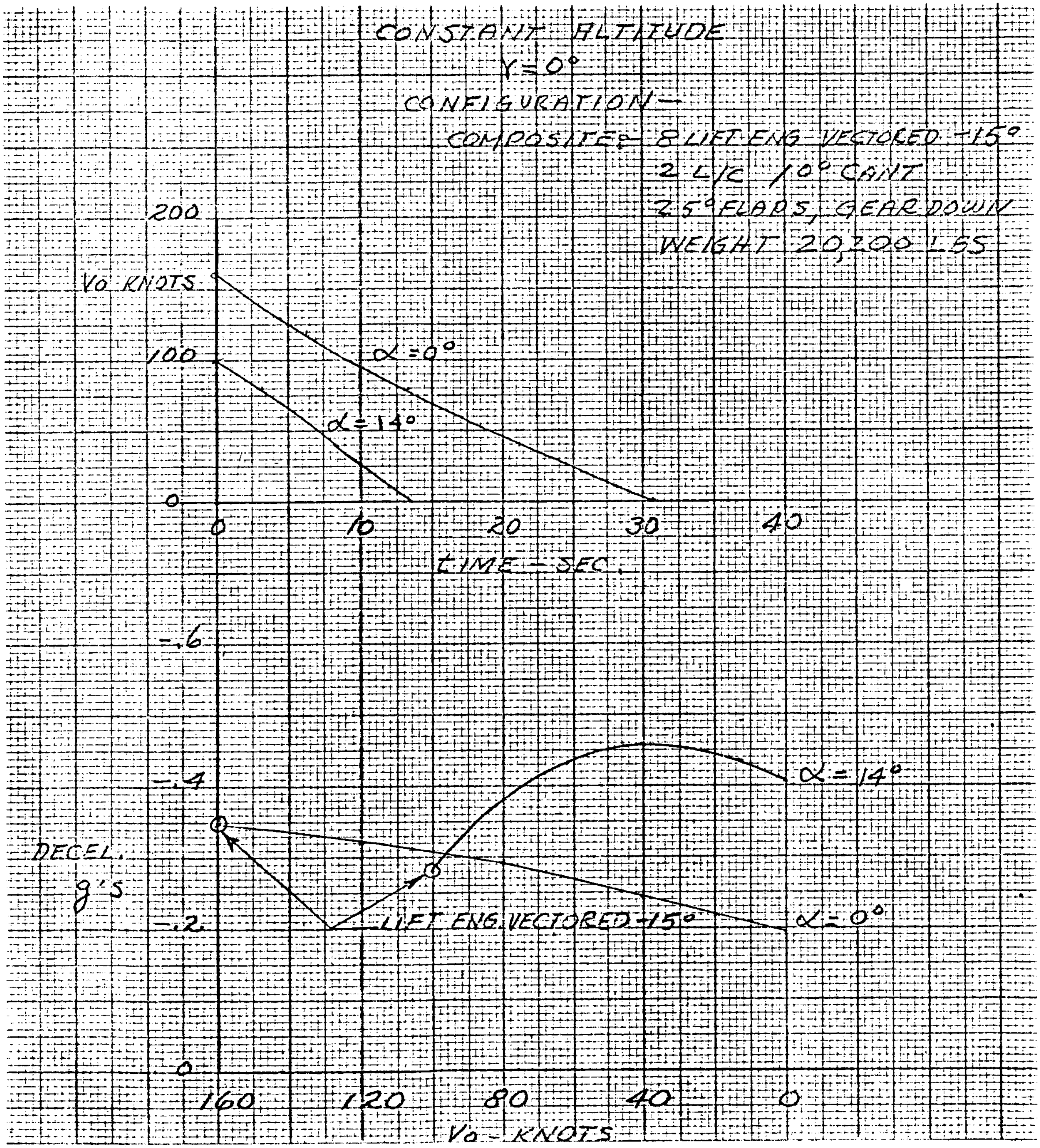


FIGURE 2-18. DECELERATION TRANSITION PERFORMANCE MOD T-39A

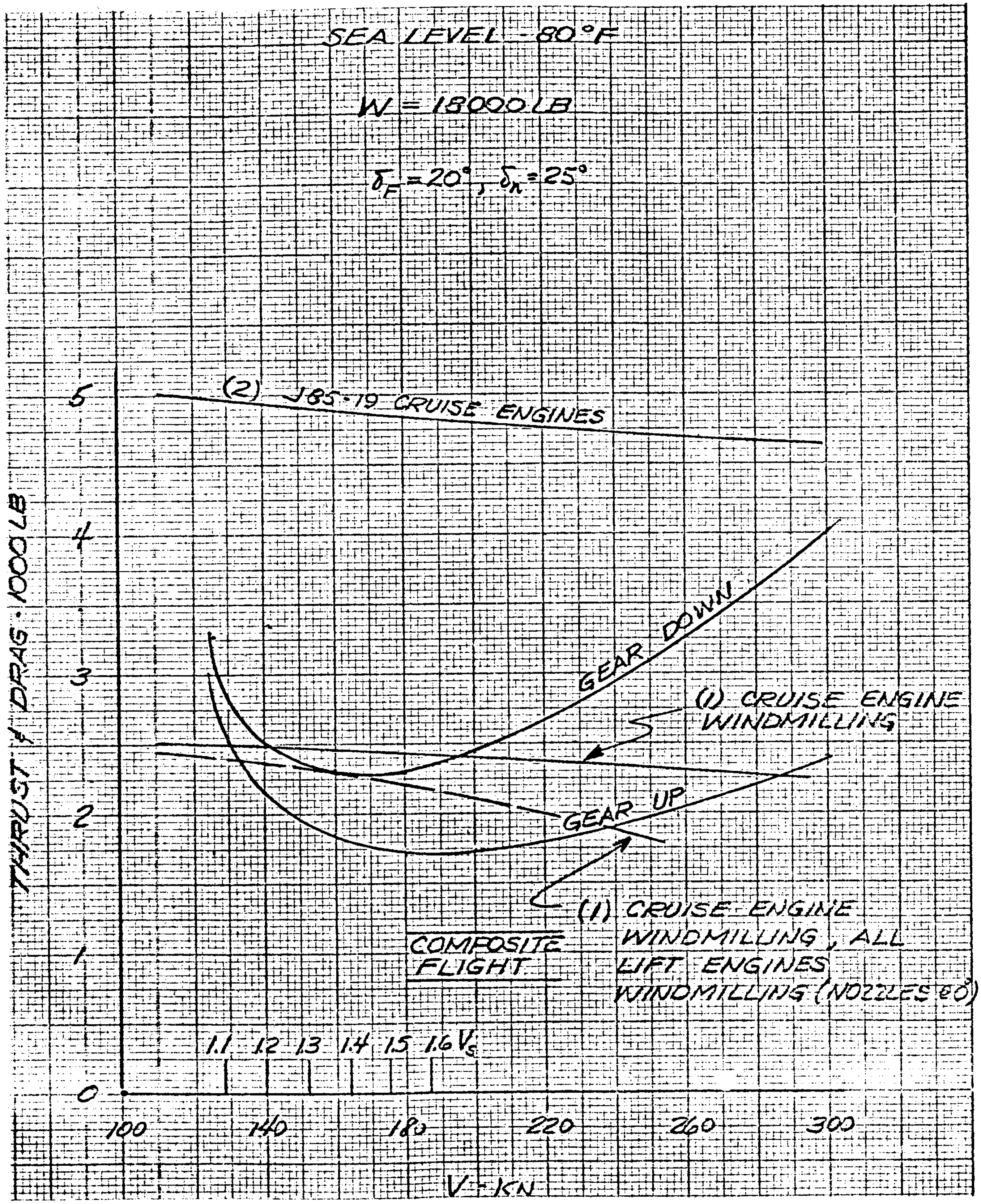


FIGURE 2-19. N-309 THRUST REQUIRED AND AVAILABLE

Composite operating was chosen because the airplane weighs 18,000 pounds as compared to 16,300 pounds in direct lift. If a cruise engine fails while lift engines are windmilling, landing gear retraction will allow excess thrust for level flight or climb. Should there be a cruise engine failure with lift engines at flight idle, the lift engines can be accelerated to maximum thrust and conversion completed, or they can be shut down and a conventional landing made. If some lift engines are only partially through the start cycle when a cruise engine fails (i.e., four engines at start and three at windmilling rpm), the start cycle can be interrupted and rpm reduced to windmill speed in two or three seconds. Then a conventional landing can be negotiated.

The modified T-39A is shown for the direct lift case at 20,000 pounds, with approach flaps and unvectored lift engines in Figure 2-20. This is the critical configuration because of the added momentum drag of two additional lift engines over the composite lift case. During an emergency the N-309 procedures also apply to the T-39A. The T-39A, however, has slightly less excess thrust when a cruise engine fails.

Conversion, for both aircraft, should be initiated at about 1.5 V_{STALL} to take advantage of maximum excess thrust at a stable flight condition in case of a cruise engine failure.

2.3.6 Stall Speeds (Conventional Flight)

Level flight stall speeds for clean, approach and landing N-309 and T-39A configurations are presented in Table 2-3. Speeds are for airplane design gross weight at sea level standard and 80^{o}F conditions.

The N-309, with a 85 psf wing loading at design gross weight, essentially meets NASA stall speed requirements and the 40 knot increment between clean and landing speeds.

No modifications are made to the T-39A wing and high lift system. At the design gross weight wing loading of 58 psf, landing stall speeds satisfy NASA requirements, but clean stall speeds are 10 knots or so lower than desired. This allows only a 30 knot stall speed spread.

2.3.7 Take-Off Distance

It is recommended that conventional take-offs with the N-309 and T-39A be made without the aid of trailing edge flaps. With approach or landing flaps, both air-

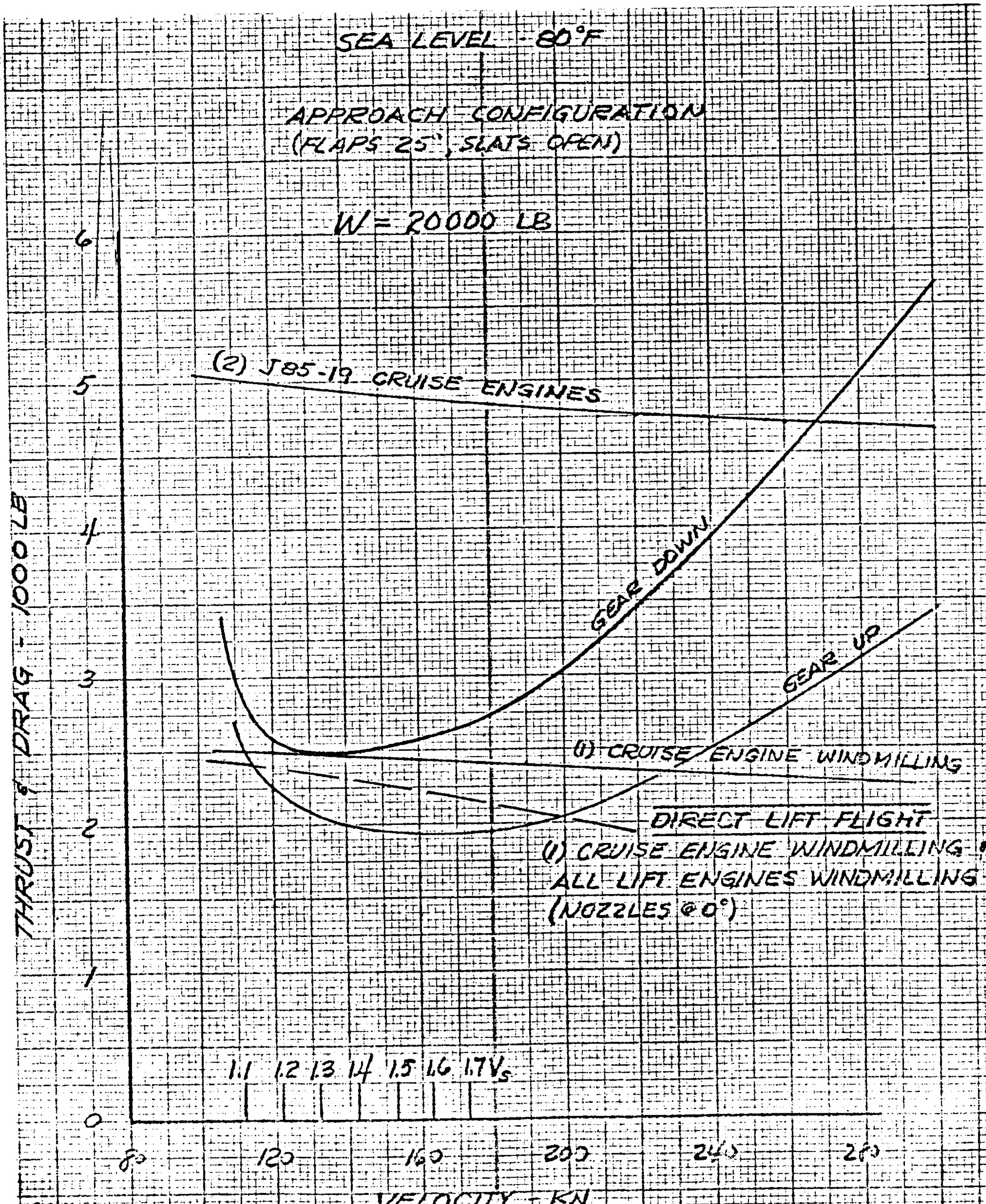


FIGURE 2-20. T-39A (MOD.) THRUST REQUIRED AND AVAILABLE

craft cannot continue a lift-off or climbout with one engine out. At design gross weight, with the aid of leading edge flaps or slats only, both aircraft can clear a 50 foot obstacle within 6,500 feet on a sea level 80^{o}F day. Take-off distance versus airplane weight is presented for each of the vehicles in Figures 2-21 and 2-22 for sea level standard and 80^{o}F conditions.

N-309 ground roll and air distances in Figure 2-21 were computed from the following relations. Ground roll is evaluated from:

$$R_{GR} = \frac{840 \dfrac{W^2}{C_{L_{TO}} S \sigma}}{64.4 \left[(F_o - \mu_R W) - F_{xTO} \right]} \ln \frac{(F_o - \mu_R W)}{F_{X_{TO}}}$$

where:

$$F_{X_{TO}} = F_{TO} - \frac{W}{C_{L_{TO}}} \left[C_{D_R} + \mu_R (C_{L_{TO}} - C_{L_R}) \right]$$

$$(\mu_R = .025)$$

Air distance is the sum of the distances required for level flight acceleration from takeoff speed, V_{TO}, to climb over 50 foot obstacle speed V_c, and the distance required to clear the obstacle at V_c.

$$R_{A_1} = \frac{2.853W \left(V_C^{\ 2} - V_{TO}^{\ 2}\right)}{64.4 \left(F_{TO} - \dfrac{C_{D_{TO}}}{C_{L_{TO}}} W - F_c + \dfrac{C_{D_c}}{C_{L_c}} W\right)} \ln \frac{F_{TO} - \dfrac{C_{D_{TO}}}{C_{L_{TO}}} W}{F_c - \dfrac{C_{D_c}}{C_{L_c}} W}$$

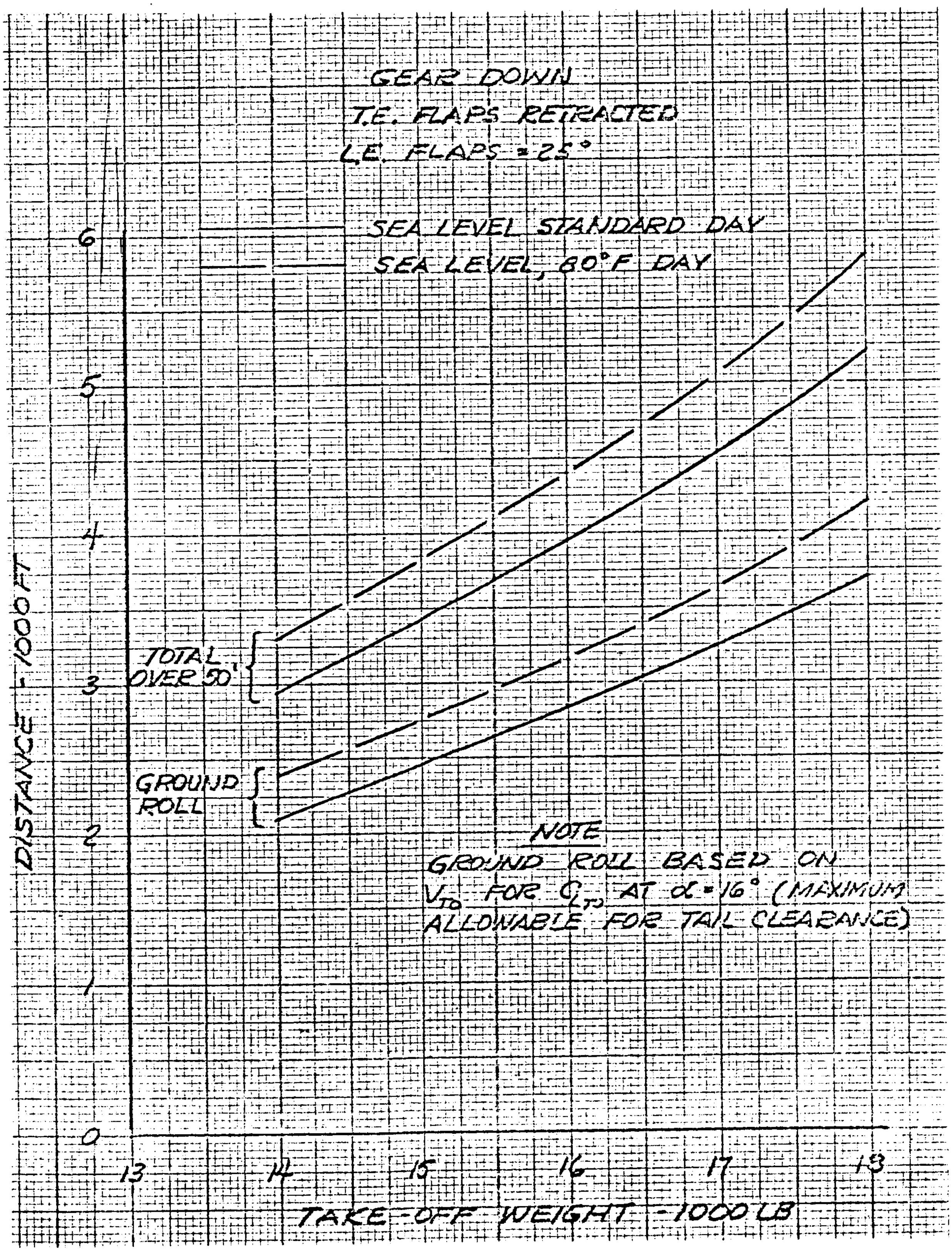


FIGURE 2-21. N-309 TAKE-OFF DISTANCE

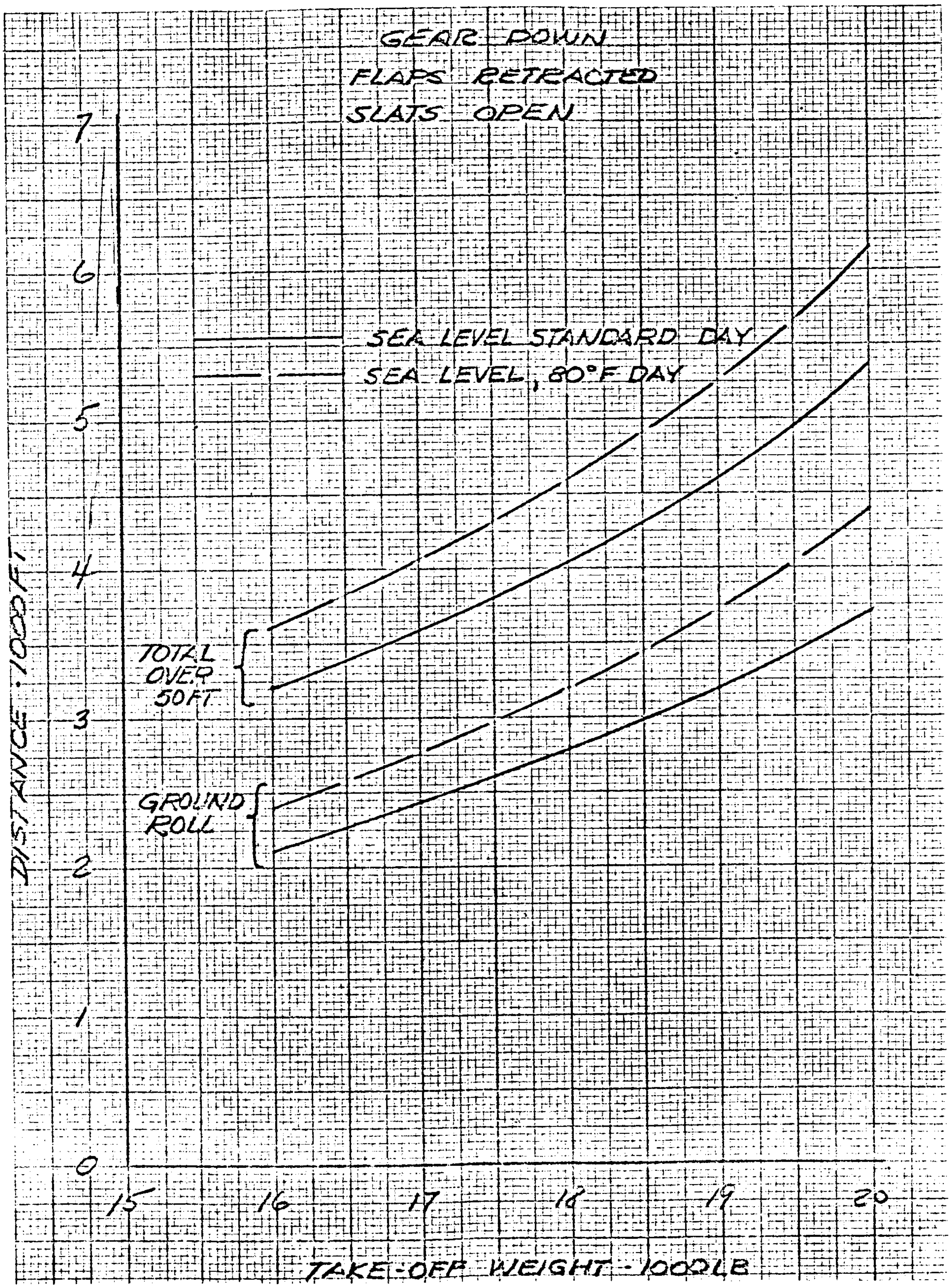


FIGURE 2-22. MOD. T-39A TAKE-OFF DISTANCE

and $$R_{A_2} = r \sin\gamma + \frac{50-h_1}{\tan\gamma} \qquad h_1 \leq 50 \text{ ft.}$$

or $$R_{A_2} = 10\sqrt{r-25} \qquad h_1 > 50 \text{ ft.}$$

where $$h_1 = r(1 - \cos\gamma)$$

$$\gamma = \sin^{-1}\left(\frac{F_c - D_c}{W}\right)$$

$$r = \frac{2.853 V_c^{\,2}}{32.2\left(\frac{C_{L_{max}}}{C_{Lc}} - 1\right)}$$

Lift off speed for the N-309 is $1.15V_{STALL}$ with trailing edge flaps retracted, because the airplane is limited by a tail clearance angle of 16° at rotation. Climb out speed for over 50 feet is $1.2V_{STALL}$.

Mod T-39A take-off distances on Figure 2-22 were obtained from the pilot's flight manual, Reference 12. The performance information in this report is based on NAA flight test data correlations.

2.3.8 Climb

Military power rate of climb on an ARDC standard day for the N-309 and T-39A, as a function of weight and altitude, are in Figures 2-23 and 2-24, respectively. These curves are instantaneous rates of climb (no climb acceleration) calculated from the following equation.

$$C = \frac{101.3\, V(F-D)}{W}$$

N-309 climb speed varies from about 350 knots at sea level to 400 knots at 1500 feet and above to the ceiling near 40,000 feet.

The T-39A has a climb speed of 300 knots at sea level to 350 knots at 15,000 and up, to the ceiling.

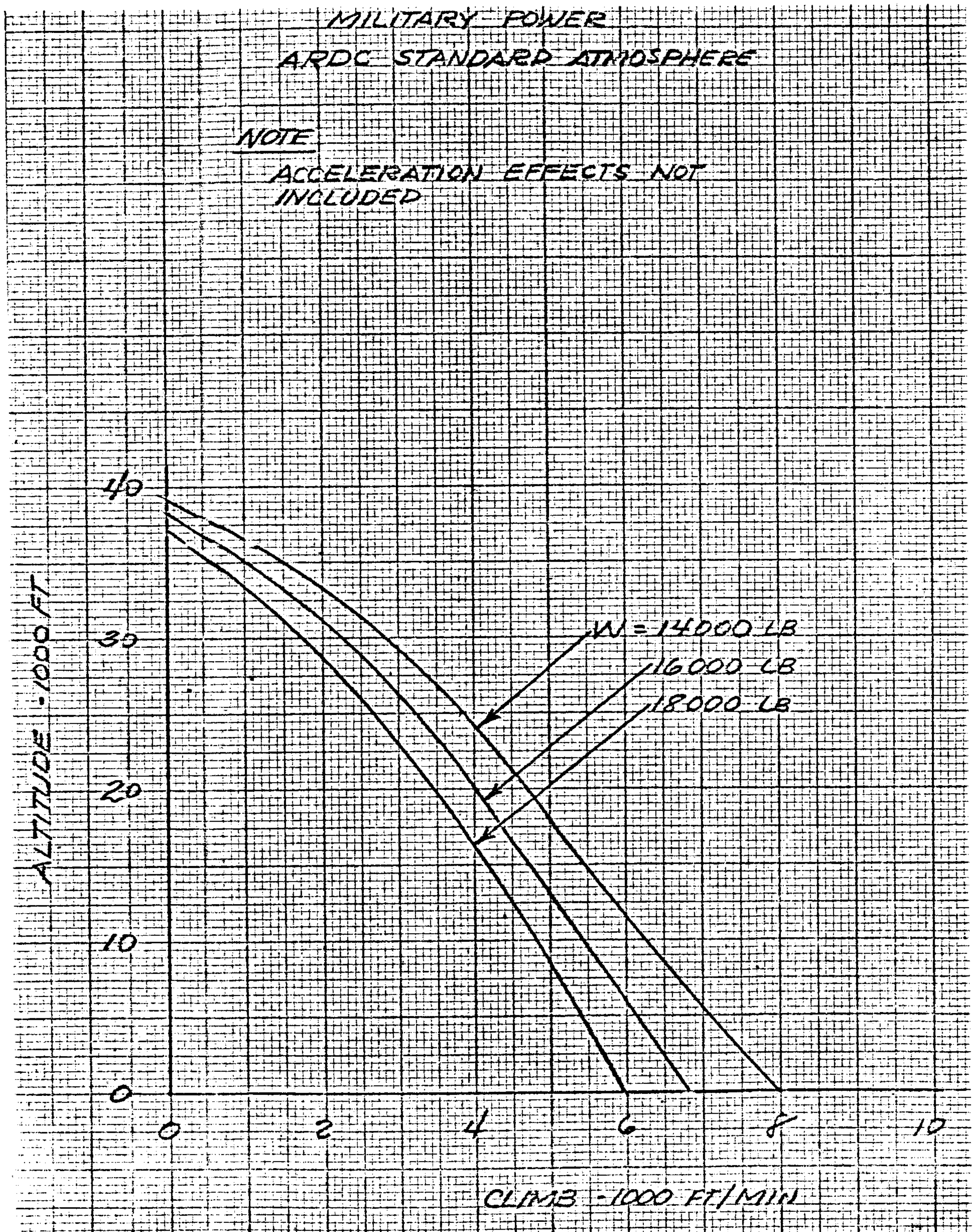


FIGURE 2-23. N-309 CLIMB

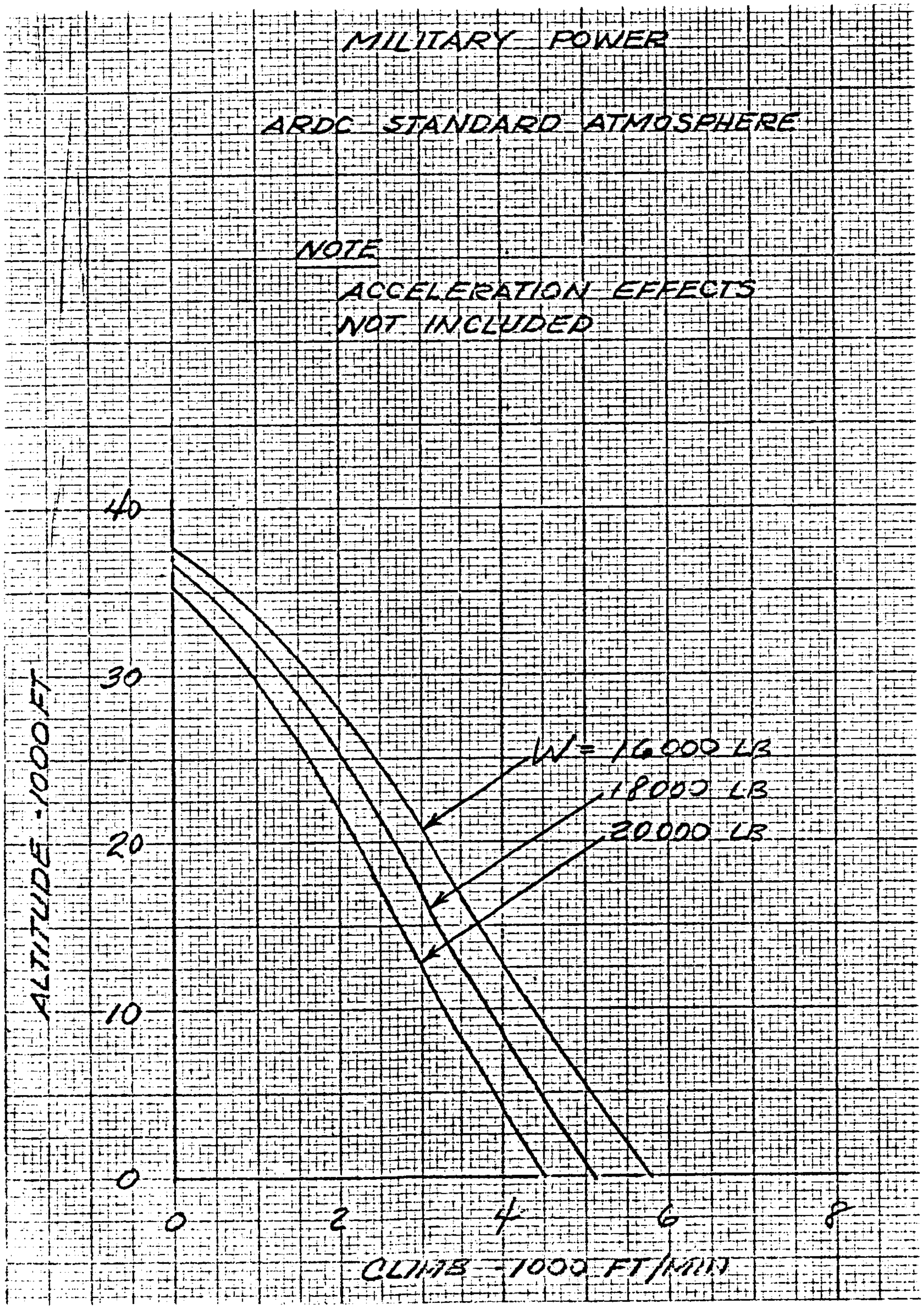


FIGURE 2-24. MOD. T-39A CLIMB

2.3.9 Speed Altitude Summary

N-309 and T-39A military power speed altitude summaries in Figures 2-25 and 2-26 show that the airplanes have comparable maximum speed and altitude capabilities. Both aircraft have maximum speeds of M = 0.8 or better and ceilings above 35,000 feet.

2.3.10 Ceilings

Forward flight and hover ceilings in Figures 2-27 and 2-28 are presented for the ARDC standard atmosphere and sea level 80°F conditions (with standard day thrust lapse rate with altitude), respectively. Hover absolute ceilings are shown for composite and direct lift operation and absolute and service ceilings for forward flight. An engine operating limit at 10,000 feet at zero speed restricts both aircraft to that maximum hover altitude. The N-309 ceilings reflect a 4% lift loss, and 6% for the T-39A, to account for a free air jet induced lift losses.

2.3.11 Landing

Conventional landing performance estimates include ground roll and air distances over a 50-foot obstacle for ARDC sea level standard and 80°F day conditions.

The N-309 landing configuration is leading edge flaps 25° and trailing edge flaps 40°, and gear down. Figure 2-29 shows total distance over a 50-foot obstacle is 5,200 feet or less, and ground roll is 3,200 feet or less at design gross weight. Distances were estimated from the following equations.

Air Distance:

$$R_{A_o} = \frac{50}{\frac{C_{D_G}}{C_{L_G}} - \frac{F_A}{W}} + \frac{\left(\frac{26.1W}{\sigma S}\right)\left(\frac{1}{C_{L_G}} - \frac{1}{C_{L_{TD}}}\right)}{\frac{C_{D_G}}{C_{L_G}} + \frac{C_{D_{TP}}}{C_{L_{TD}}} - \frac{F_A + F_I}{W}}$$

$$F_A = \text{approach thrust} = \frac{-(R/S)W}{1.689V_G} + D$$

$$F_I = \text{idle thrust}$$

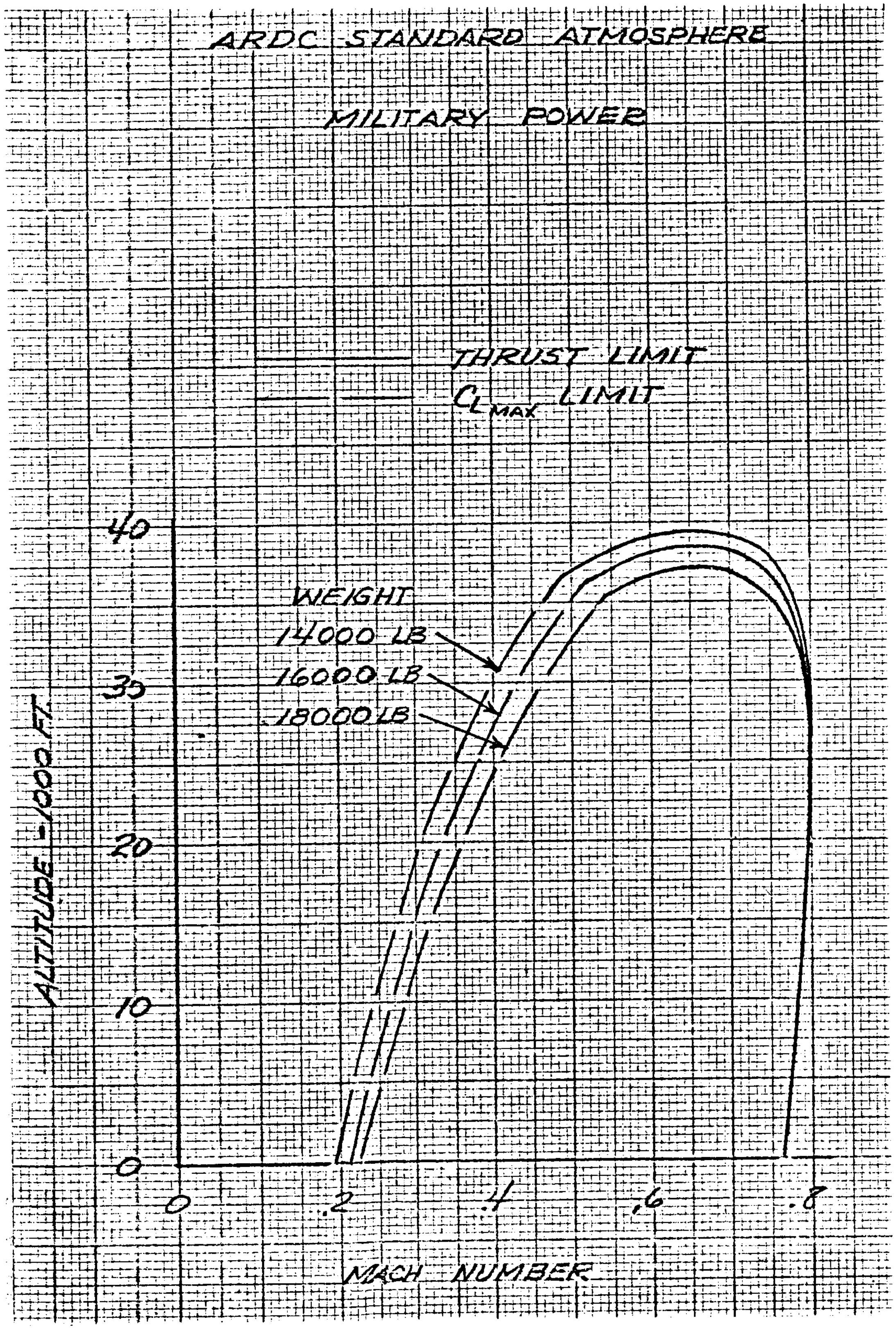


FIGURE 2-25. N-309 MACH NUMBER SUMMARY

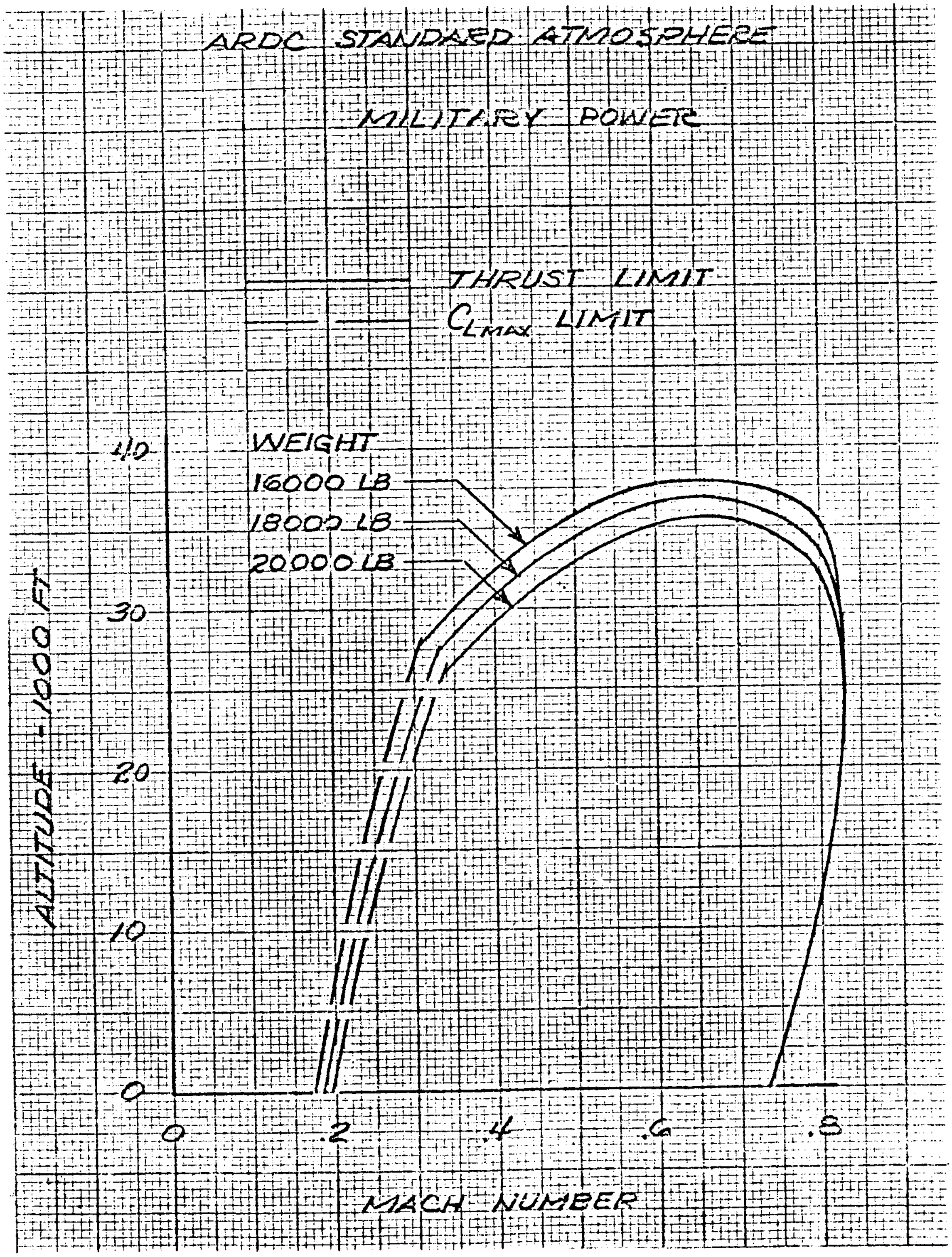


FIGURE 2-26. MOD. T-39A MACH NUMBER SUMMARY

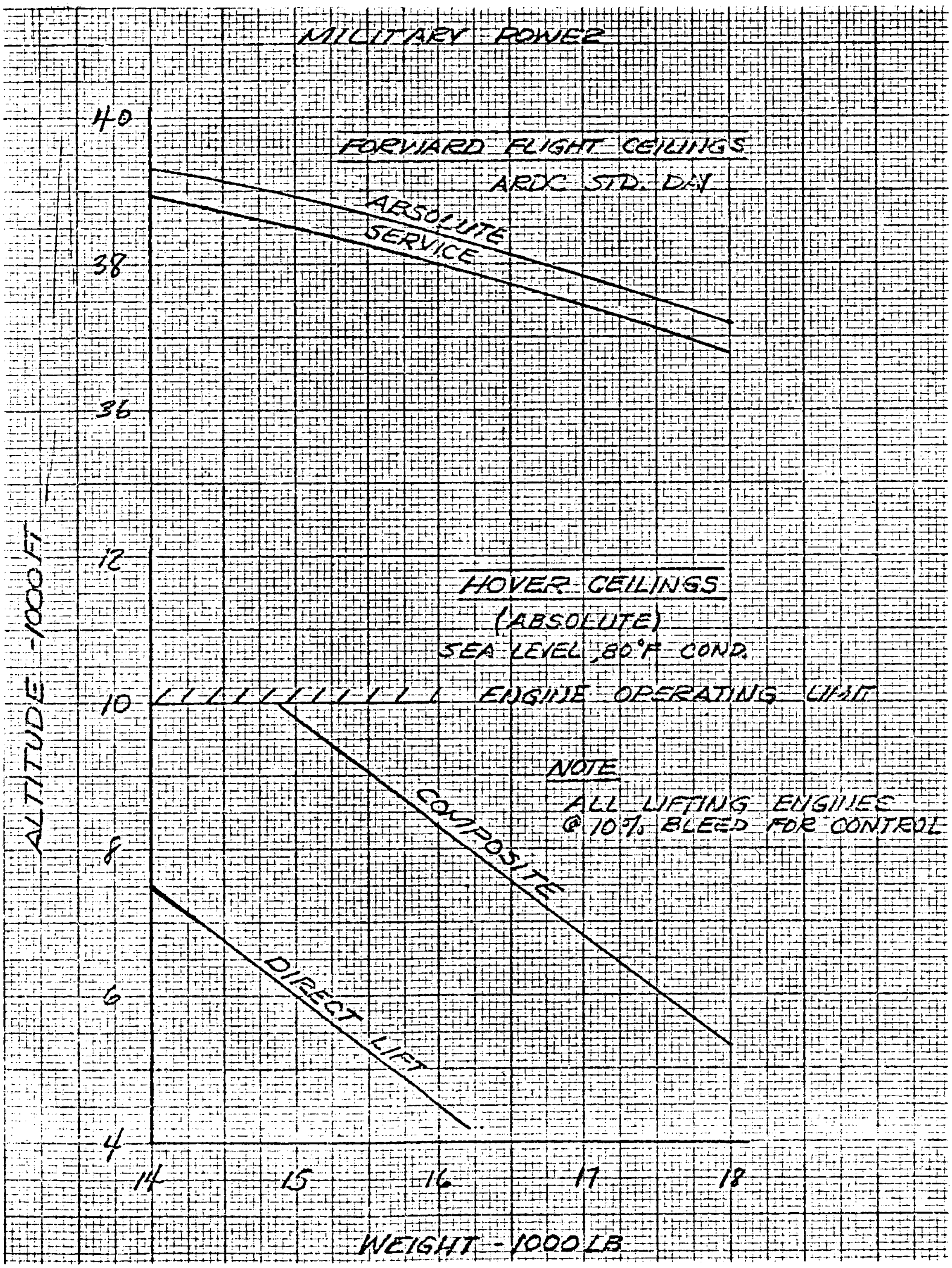


FIGURE 2-27. N-309 CEILINGS

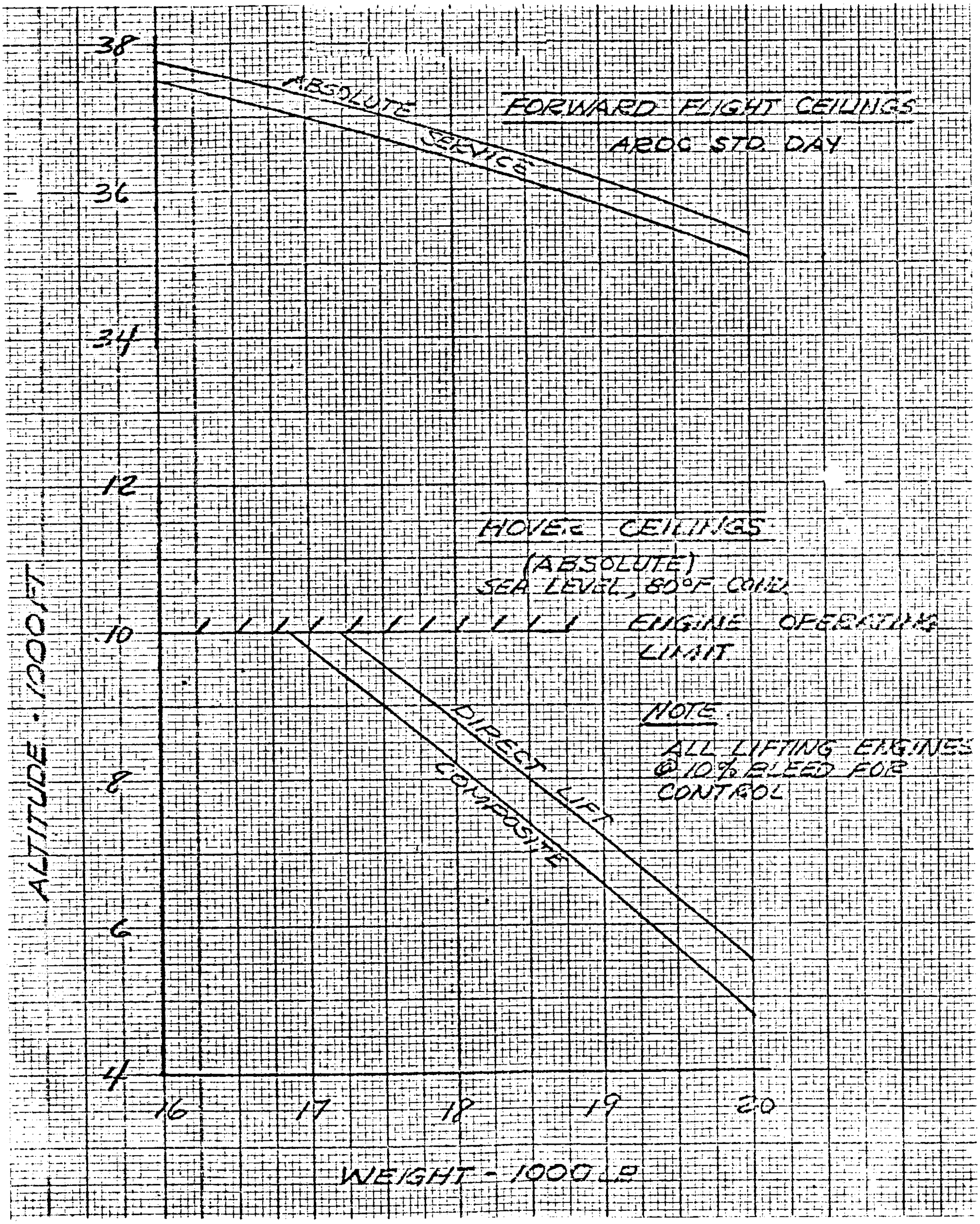


FIGURE 2-28. MOD. T-39A CEILINGS

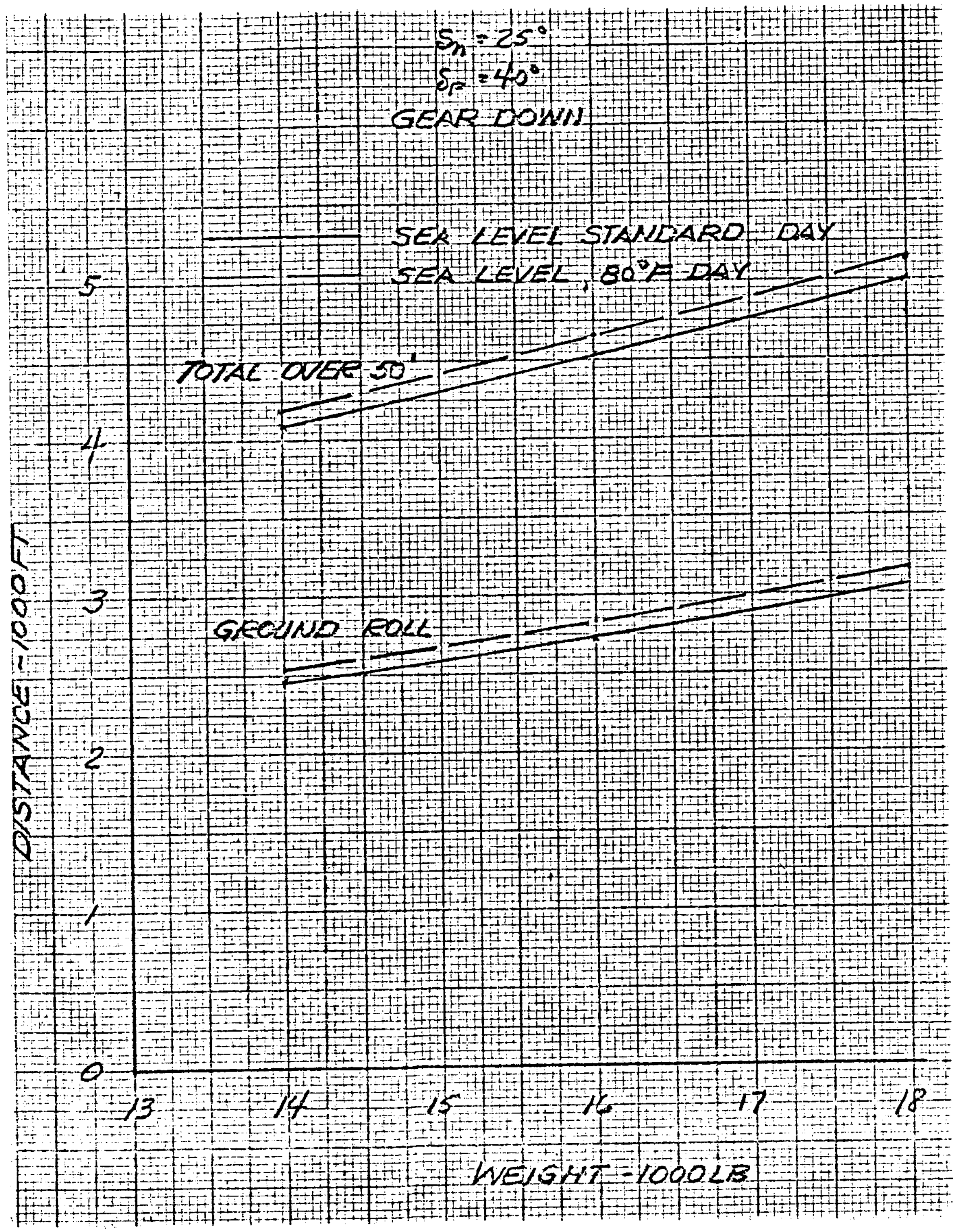


FIGURE 2-29. N-309 LANDING DISTANCE

Free Roll:

$$R_{FR} = 3(1.689)\ V_{TD}$$

Ground distance is considered the sum of a three second free roll distance (above) and a braked distance defined below.

Braked Distance:

$$R_B = \frac{2.853\, C_{L_{TD}}\, V_{TD}^{\ 2}}{64.4\left(\mu_B C_{L_B} - C_{D_B}\right)} \ln \frac{\mu_B - \frac{F_I}{W}}{\frac{C_{D_B}}{C_{L_{TD}}} + \mu_B\left(1 - \frac{C_{L_B}}{C_{L_{TD}}}\right) - \frac{F_I}{W}}$$

if $\mu_B C_{L_B} - C_{D_B} = 0$

then $R_B = \dfrac{2.853\ V_{TD}^{\ 2}}{64.4\left(\mu_B - \frac{F_I}{W}\right)}$ $\qquad \mu_B = .30$

T-39A landing distance in Figure 2-30 was obtained from Reference 2-12. It is based on flight test data correlations for the slats extended and 100% trailing edge flap (25^o) configuration. These data show a more rapid increase in landing distance with weight and temperature than estimates with the previous equations. This airplane will require a 9,000 foot runway for above normal temperatures and high gross weight landing operations.

2.3.12 Mission Performance

In addition to the basic hover mission, the Ferry and Transition missions are shown to explore airplane capabilities. Ambient conditions for the hover mission are sea level and 80^oF, and the Ferry and Transition missions are based on an ICAO standard day. Fuel consumption includes a 5% service tolerance for all missions in accordance with Reference 2-15.

The hover mission is defined as follows for composite or direct lift operation.

1. Warm-up and VTO allowance (1 minute at hover thrust).
2. Hover at T/W = 1.0 + jet interference lift loss.

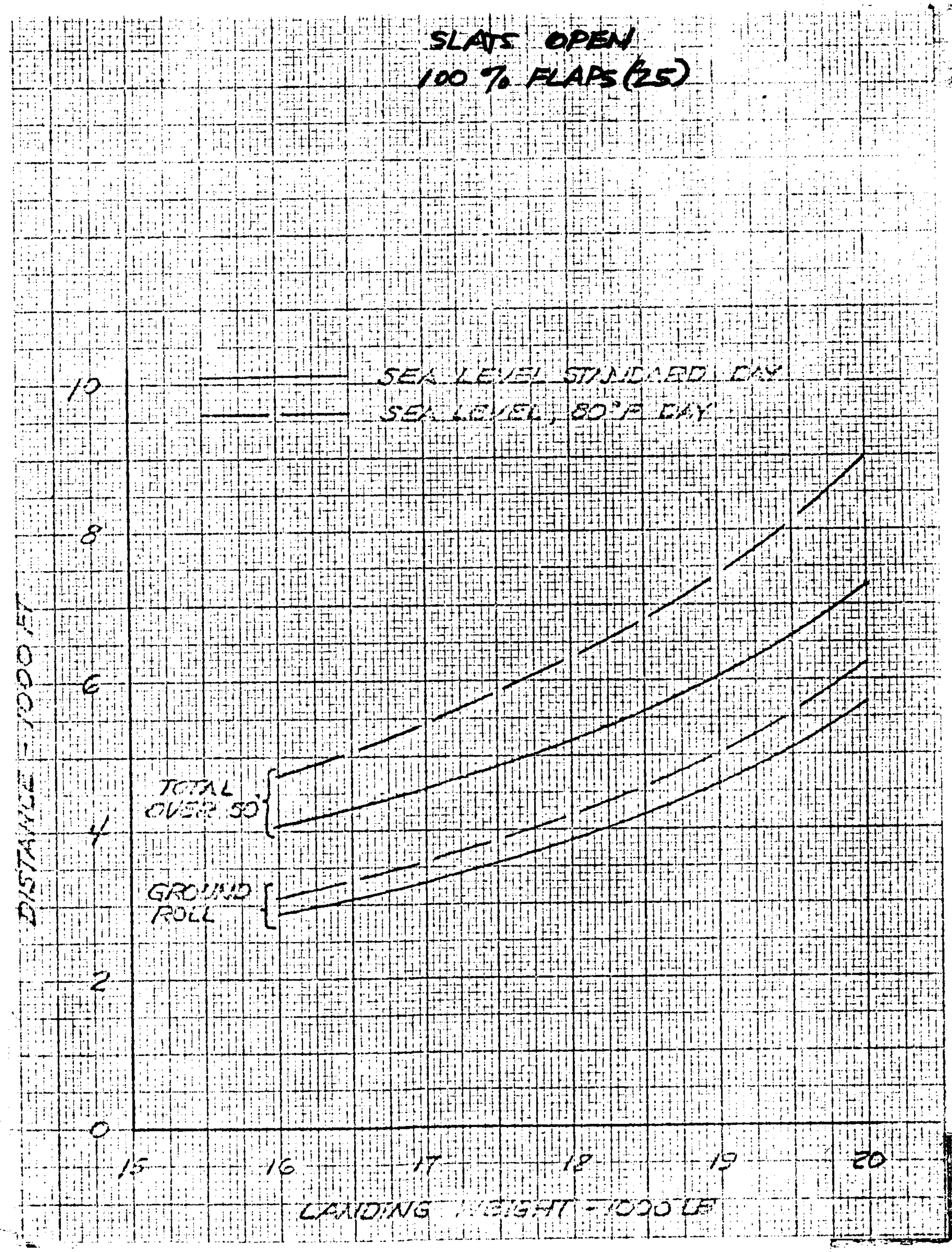


FIGURE 2-30. MOD. T-39A LANDING DISTANCE

3. Vertical landing with 5% initial fuel for reserve.

Hover times for this mission are indicated below for the N-309 and T-39A.

CONFIGURATION	N-309	T-39A
Composite	12.7 min	12.1
Direct Lift	7.6 min	10.3

Transition Mission

1. Warm up and VTO, (1 minute at hover thrust).
2. Transition at military thrust from V = 0 to 1.5 V_{stall} (composite flight) and convert to level flight.
3. Accelerate to V_{climb} and Climb to 25,000 feet at military thrust.
4. Descend to sea level at V=300 kn and decelerate to 1.5 V_{stall}.
5. Convert to VTOL configuration (composite) and decelerate to V = 0.
6. Land with 5% initial fuel for reserve plus fuel for 20 minutes loiter at sea level at speed for maximum endurance. (MIL-C-5011A reserves, Reference 15).

The N-309, with internal fuel, can perform the foregoing mission (items 2 and 5) three times, landing the last time only. With full internal fuel the T-39A can complete the mission twice and hover for about 1.5 minutes before landing.

Ferry Mission

1. Warm up and take-off (5 minutes at normal power).
2. Climb to 36,089 feet at military thrust.
3. Cruise at V for maximum range.
4. Loiter at sea level at V for maximum endurance.
5. For 20 minutes.
6. Land with 5% initial fuel for reserve.

This is a general purpose or escort fighter ferry mission specified by MIL-C-5011A. Mission range including 84 NMi for climb is 702 NMi for the N-309. T-39A range is 609 NMi which includes 123 NMi climb range.

2.3.13 Hover Endurance Sensitivity

The effect of a ± 50 percent change in engine installation and free air jet induced lift losses on hover endurance are presented in Figure 2-31. An increase in installation losses reduces total thrust and fuel must be off-loaded. Conversely, thrust gains allow for additional fuel. The airplane has volume for the extra fuel. Hover time is also affected by engine SFC when installation losses cause thrust changes. Hover time increments, in Figure 2-31, reflect the fuel volume and SFC effects of changes in engine installation losses. The jet induced lift loss does not constitute an engine installation loss change; therefore, the time increment is due to fuel loading only.

2.4 STABILITY AND CONTROL

2.4.1 Reaction Control - General

The "nominal" control powers required for hovering flight were calculated in accordance with the requirements of paragraphs 2.12, 3.1 and 3.12 of AGARD 408, modified as per NASA instructions. The NASA modifications comprise factoring the AGARD 408 specified airplane angular response values by 1.5, 2.0 and 1.5 for pitch, roll and yaw respectively. The resulting control powers assumed a step input of full control and the AGARD 408 rate damping. Typically the equation of motion is of the form;

$$\text{Reqd} \qquad M_c = \frac{I_y \, \theta_{t=1}}{\tau^2 \left(e^{-1/\tau} + \frac{1}{\tau} - 1\right)}$$

$$\text{where} \qquad \tau = -\frac{-I_y}{M_{\dot{\theta}}} \qquad M_{\dot{\theta}} = \frac{\partial M}{\partial \dot{\theta}} \quad \text{ft. lb/rad/sec}$$

Roll and yaw equations are similar.

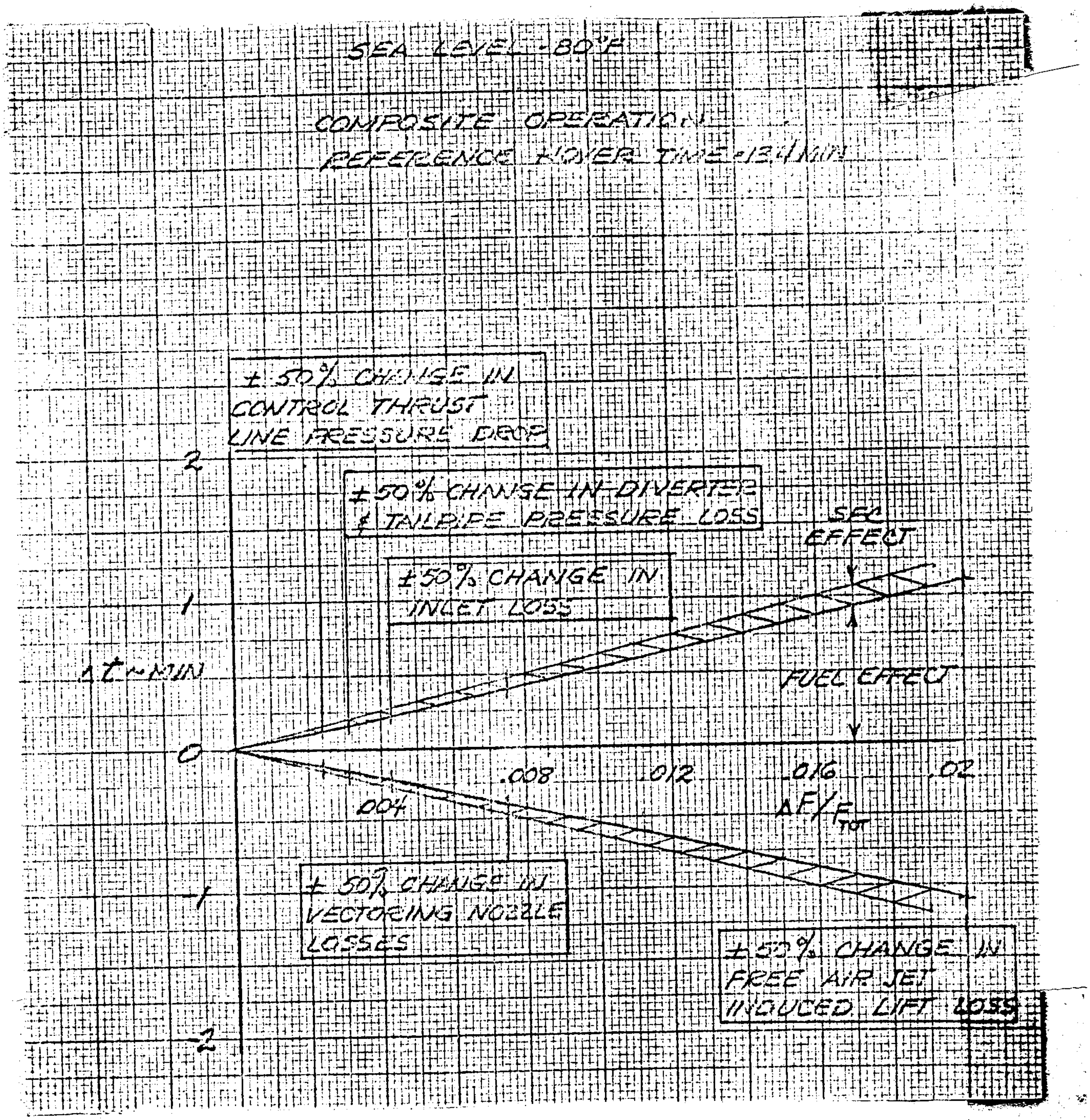


FIGURE 2-31. N-309 HOVER ENDURANCE SENSITIVITY

The design requirements under normal conditions with all engines operating are shown in the following table.

Axis	Response for full control input (degrees in first sec)	Damping (ft. lb./rad./sec.)
Longitudinal	$\frac{450}{(W+1000)^{1/3}}$	$15(I_y)^{0.7}$
Lateral	$\frac{600}{(W+1000)^{1/3}}$	$25(I_x)^{0.7}$
Directional	$\frac{270}{(W+1000)^{1/3}}$	$27(I_z)^{0.7}$

In addition to complying with the above requirements, consideration must also be given to the capability to trim out an engine failure or a 35-knot crosswind and still retain some degree of maneuverability.

The control margins remaining following the trimming out of an engine failure, etc., are, as a minimum, the following:

Pitch - 20% of the "nominal" hover value
Roll - 50% of the "nominal" hover value
Yaw - 33% of the "nominal" hover value

These values are taken (or inferred) from paragraphs 5.5, 3.17, and 3.14 respectively of AGARD 408. These margins are also to be maintained through transition, this aspect being discussed in detail in Section 2.4.3 following.

2.4.2 Simultaneous Control

Simultaneous rather than 100 percent single axis control thrust requirements make the greatest demands on available engine bleed air. Comparisons between simultaneous control thrusts required and available, expressed as a function of engine thrust ratio, are shown in Figures 2-32 and 2-33 for the N-309 and Mod. T-39A respectively.

Aircraft weights shown in these figures include jet induced lift losses (OGE). The minimum weight condition is with approximately 15 percent fuel.

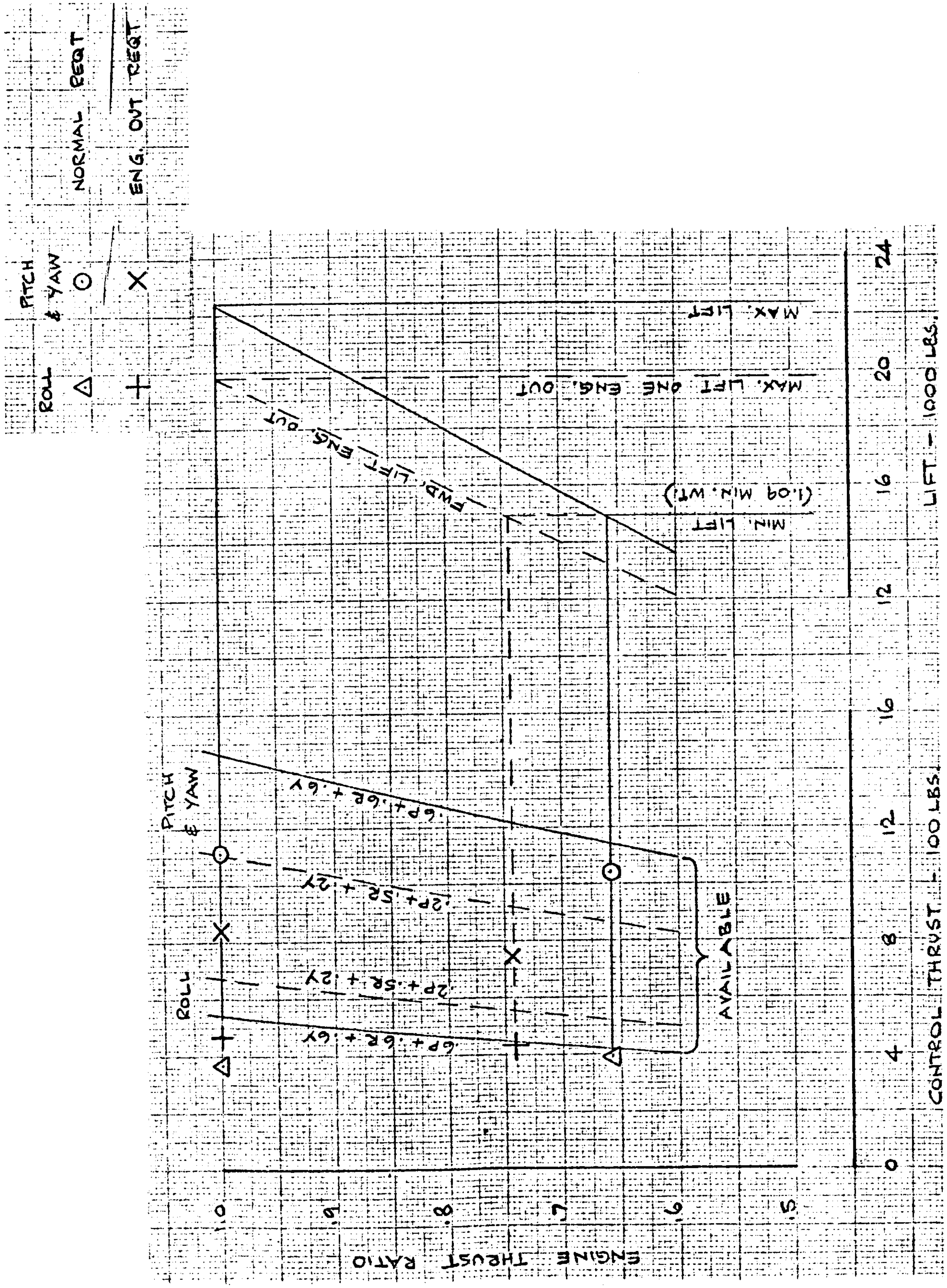


FIGURE 2-32. SIMULTANEOUS CONTROL THRUST AVAILABLE AND REQUIRED N-309 -10% BLEED

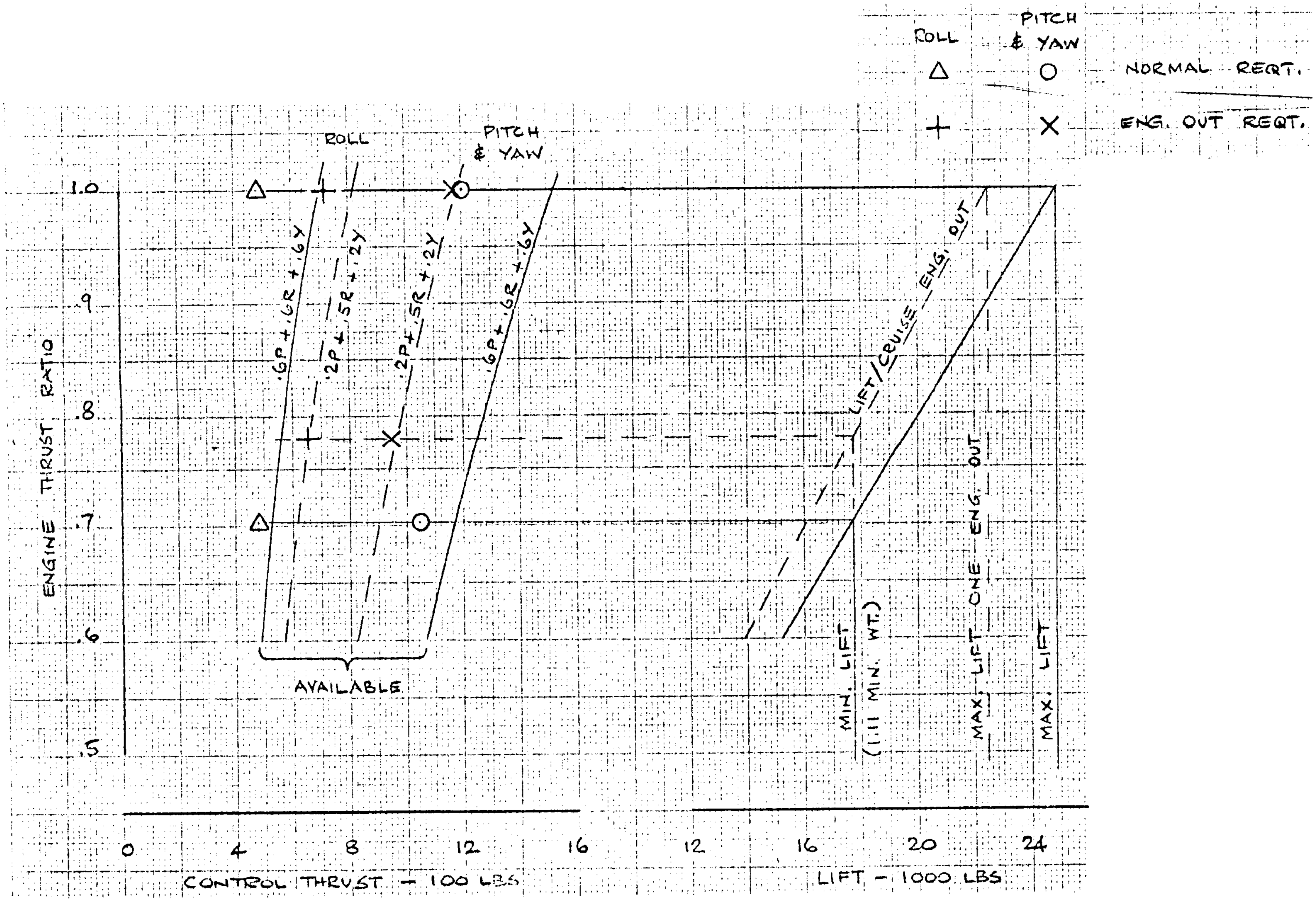


FIGURE 2-33. SIMULTANEOUS CONTROL THRUST AVAILABLE AND REQUIRED – MOD. T-39A – 10% BLEED

Normal (all engines operating) requirements are 60 percent of maximum control simultaneously on all axes, and 20 percent pitch and yaw control plus 50 percent roll control with one engine out.

Examination of these figures reveals that all normal and engine out requirements are met or exceeded for both N-309 and Mod. T-39A.

2.4.3 Transition Trim

Transition trim requirements and capability have been studied for both the N-309 and modified T-39A aircraft. Flight condition analyzed is a full throttle accelerating transition in the composite mode with the engines' thrust vectored 28° aft (maximum available). Specifically the cases studied were (for $0 \leq V \leq 180$ knots),

1. Longitudinal
   a. Normal all engine operation ~ N-309 (Figure 2-34) and Mod. T-39A (Figure 2-38).
   b. Critical engine out ~ N-309 (Figure 2-35).
2. Lateral
   Sideslip angle equivalent to a 35 knot sidewind or $\beta = 15^{\circ}$ whichever is larger ~ N-309 (Figure 2-36) and Mod. T-39A (Figure 2-39).
3. Directional
   Sideslip angle equivalent to a 35 knot sidewind or $\beta = 15^{\circ}$ whichever is larger ~ N-309 (Figure 2-37).

The figures indicated above which illustrate the trim characteristics include the following information:

Figures 2-34 and 2-35 ~ N-309 longitudinal trim (normal and critical engine out):

(1) Trim moment required which includes the contributions of,
  (a) Vectored thrust
  (b) Inlet momentum drag
  (c) Exhaust jet induced effects
  (d) Angle of attack (two angles of attack considered)
(2) Trim moments (as (1) above) plus 20% of the "nominal" hover pitch control moment, where the "nominal" control moment is as defined in

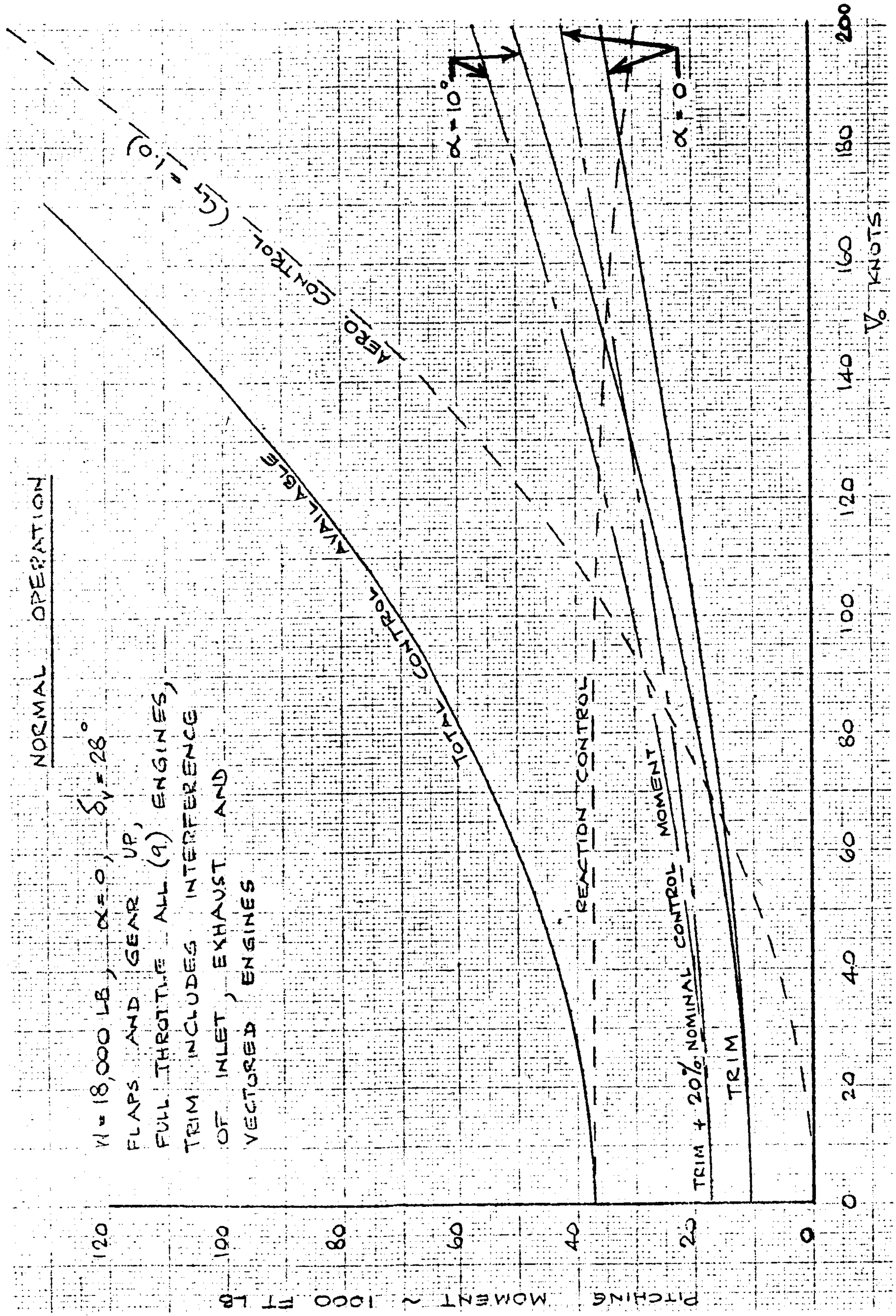


FIGURE 2-34. STABILITY AND CONTROL – TRANSITION PITCH CONTROL – N-309

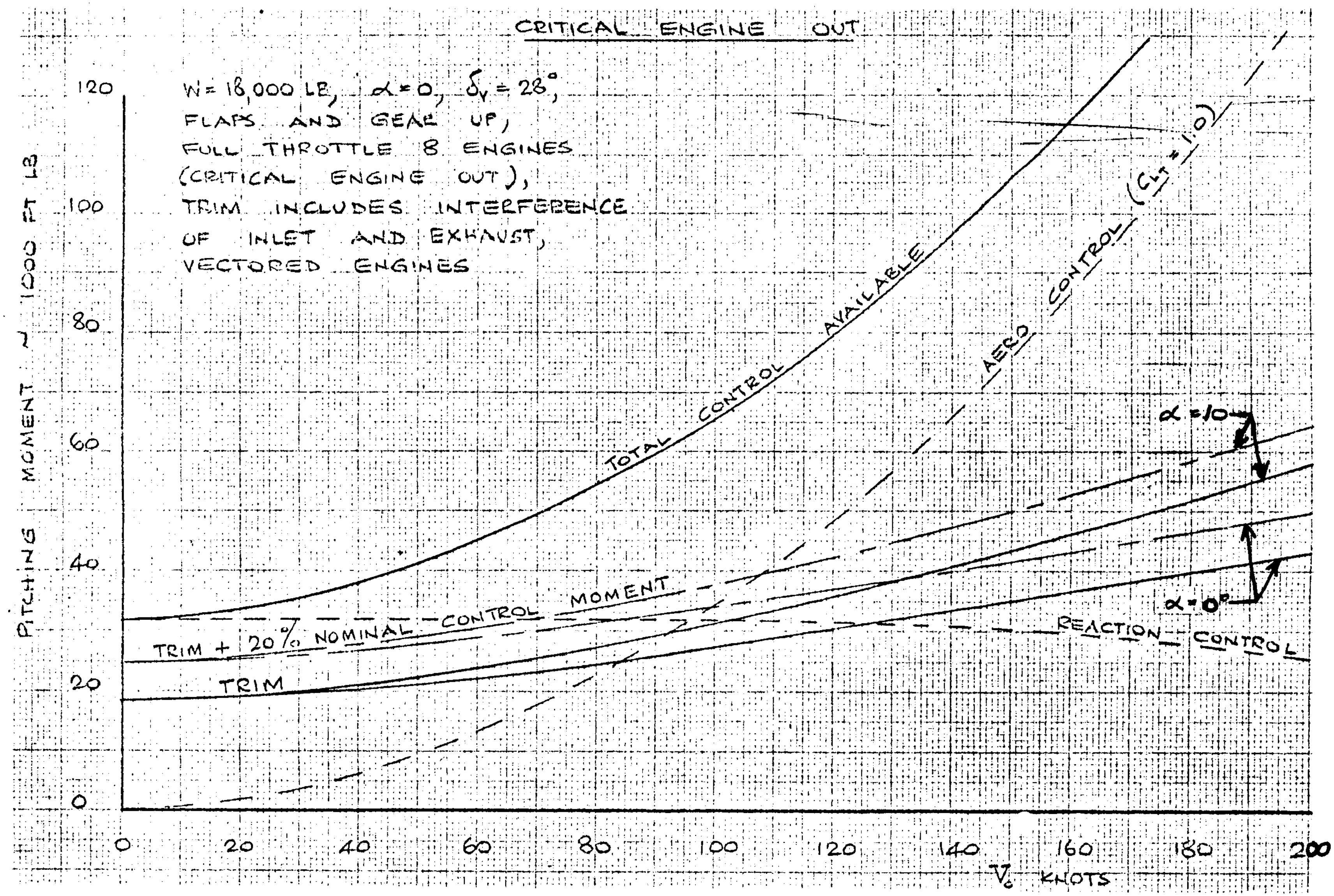


FIGURE 2-35. STABILITY AND CONTROL – TRANSITION PITCH CONTROL – N-309

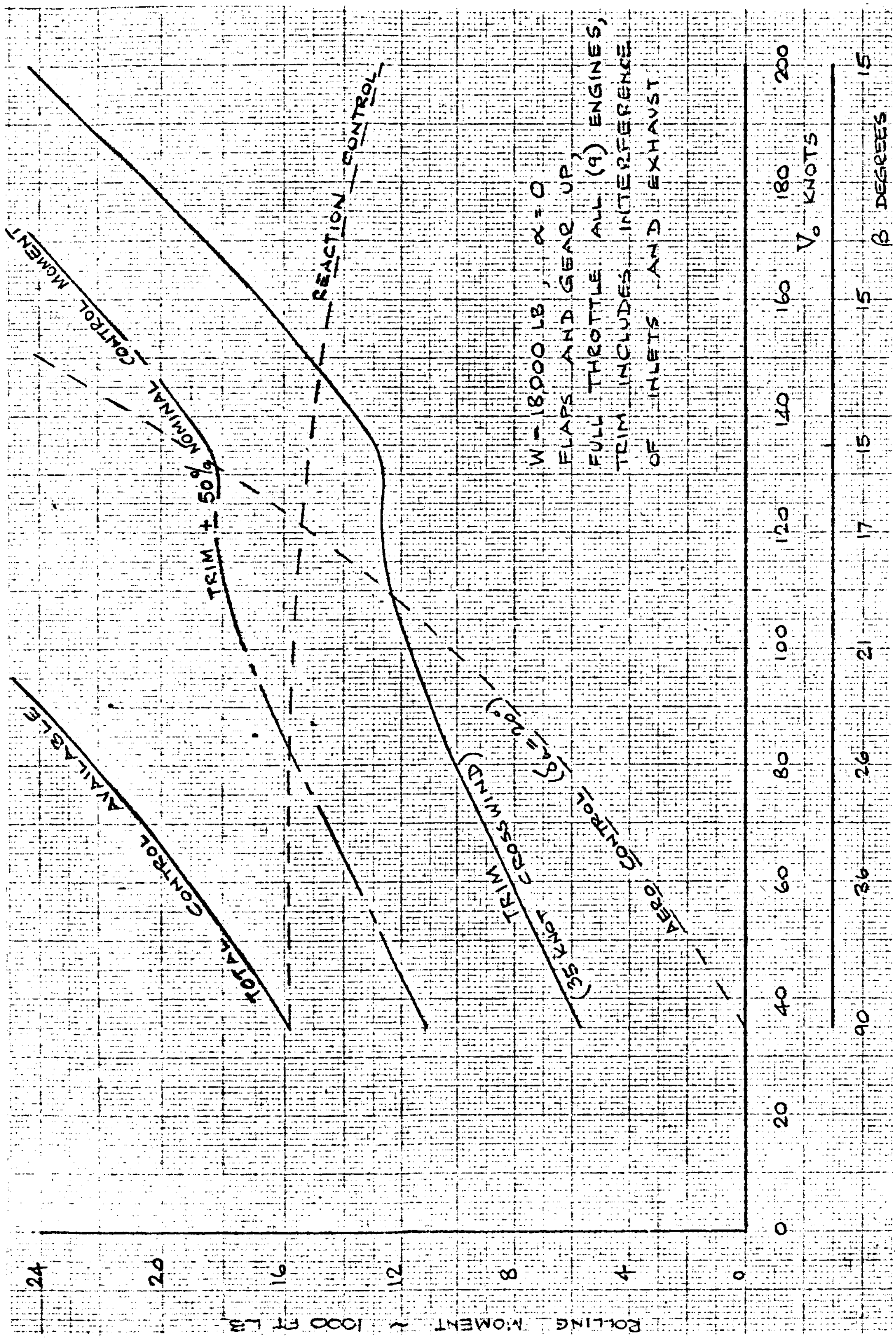


FIGURE 2-36. STABILITY AND CONTROL – TRANSITION ROLL CONTROL – N-309

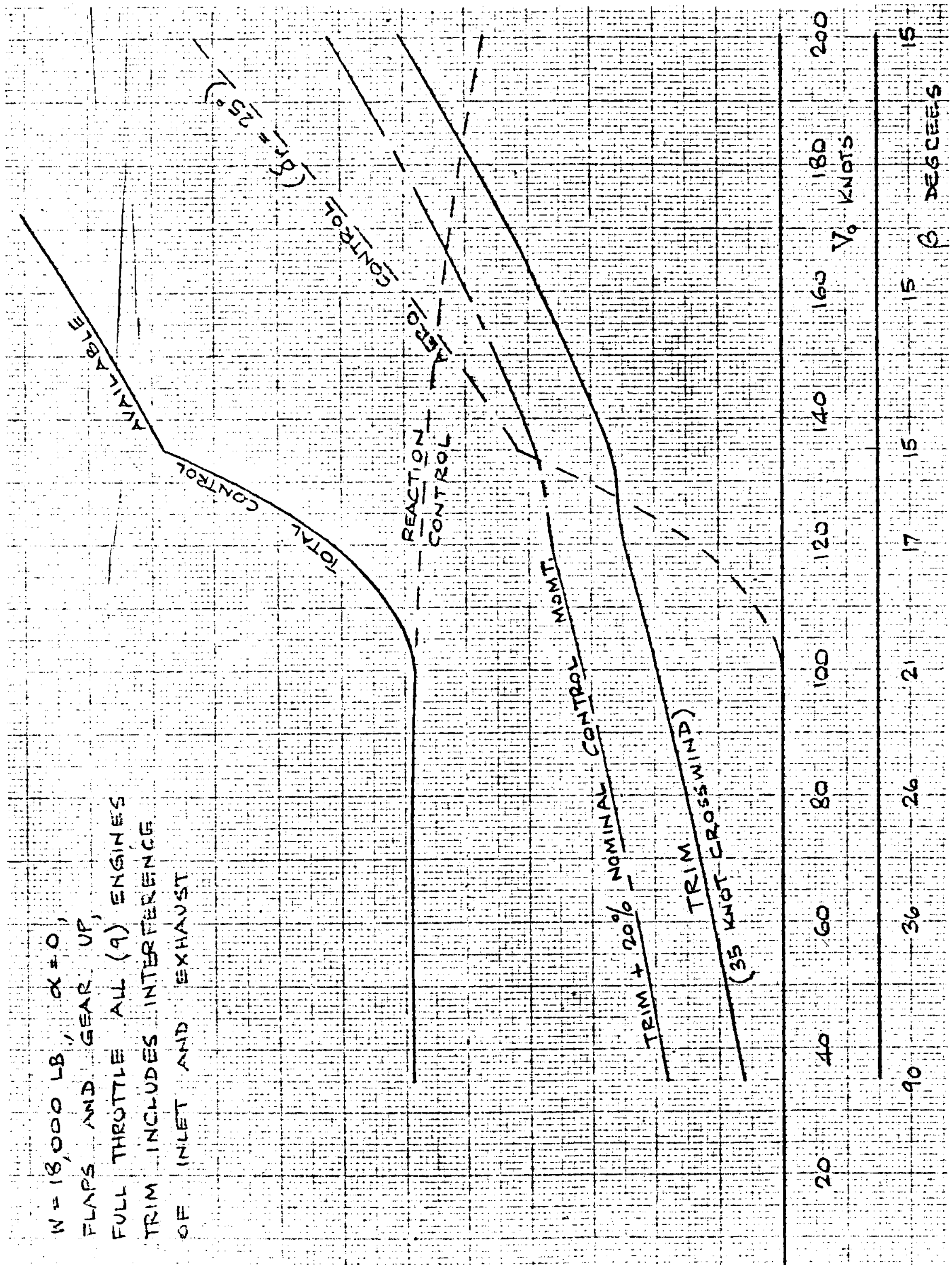


FIGURE 2-37. STABILITY AND CONTROL – TRANSITION YAW CONTROL – N-309

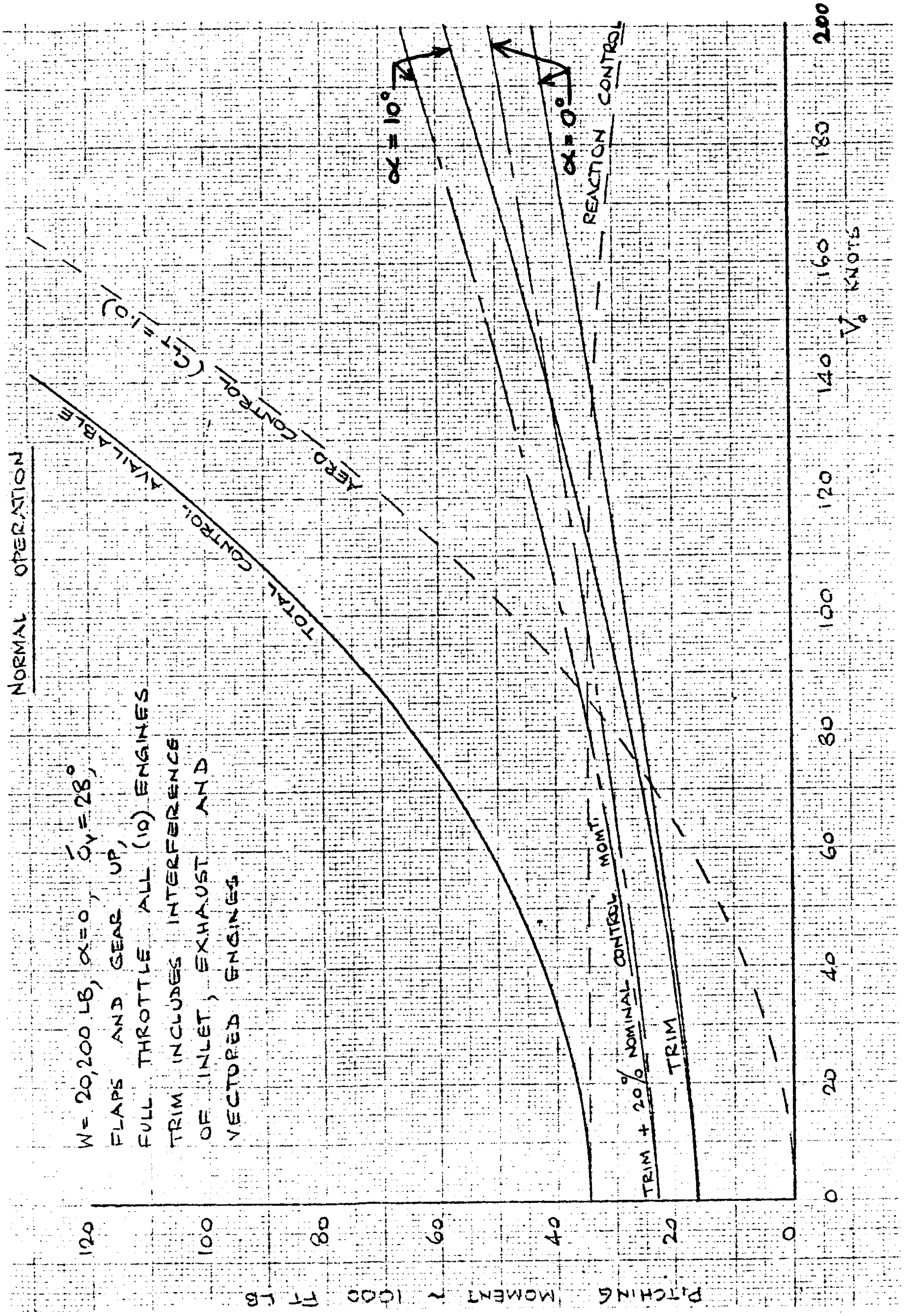


FIGURE 2-38. STABILITY AND CONTROL – TRANSITION PITCH CONTROL – MODIFIED T-39A

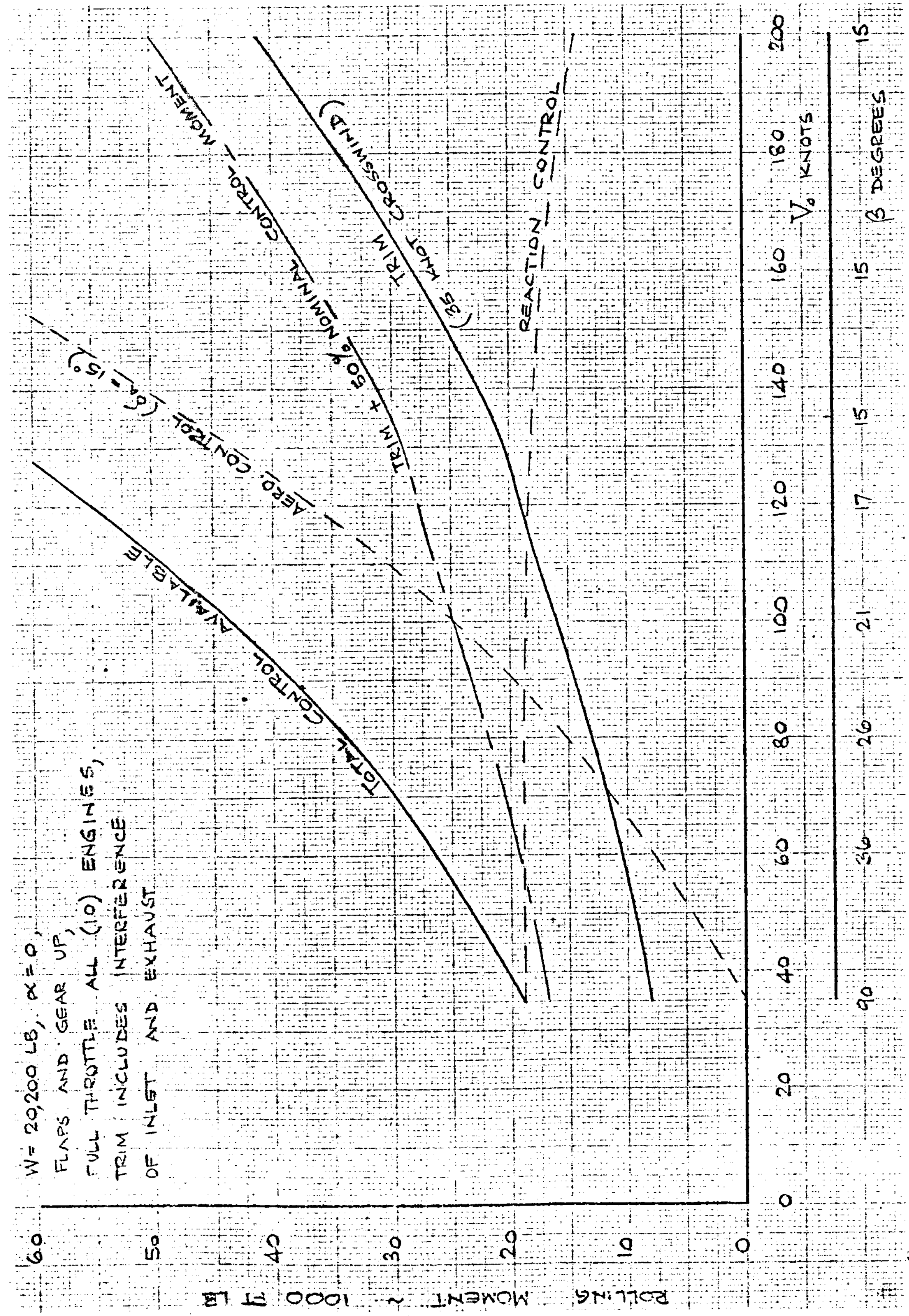


FIGURE 2-39. STABILITY AND CONTROL - TRANSITION ROLL CONTROL - MODIFIED T-39A

Section 2.4.1. This is the AGARD 408 recommended minimum margin to allow for maneuvering and disturbances (paragraph 5.5 of AGARD 408).

(3) Available reaction control moment at full power setting. The hover value is maintained constant up to 80 knots and drops to 80% of the initial value at 200 knots. This choice of control effectiveness decay was influenced by the data of Reference 2-22.

(4) Aerodynamic control (horizontal tail) assuming $C_{L_T} = 1.0$.

(5) Total control (reaction plus aerodynamic).

These Figures demonstrate that the aircraft is easily trimmed through a maximum acceleration transition with a considerable control margin remaining both for normal operation and with the critical engine failed. The maximum power accelerating transition is the most critical case since the pitching moment contributions, namely vectored thrust, jet induced effects and angle attack (when positive), all act in the same sense - nose up.

In comparison, the moment contributions in a decelerating transition are nose down due to forward vectoring of the lift engines thrust and nose up, once again due to interference effects and angle of attack.

Figures 2-36 and 2-37 ~ N-309 lateral and directional trim (normal operation):

(1) Trim moment required to balance 35 knot crosswind or 15^{o} sideslip and includes the contributions of,
  (a) Inlet momentum drag
  (b) Exhaust jet induced effects
  (c) Aerodynamic forces

(2) (a) Lateral - Trim moment plus 50 percent of the "nominal" hover roll control moment, where the "nominal" control moment is as defined in Section 2.4.1.
  (b) Directional - Trim moment plus 20 percent of the "nominal" hover yaw control where the "nominal" control moment is as defined in Section 2.4.1. These control margins are taken from paragraph 4.2.3.(b) of NASA RFP L-7151.

(3) Available reaction control moment at full power setting. Again, the hover value is maintained up to 80 knots and drops to 80 percent of the initial value at 200 knots. This control effectiveness decay is again based on the data of Reference 2-22.

(4) Aerodynamic Control

(a) Ailerons - $\delta_a = 20^\circ$

(b) Rudder - $\delta_r = 25^\circ$. The rudder contribution is derived as follows. It is assumed that for $\beta > 20^\circ$ (for $V < 100$ knots approximately) the vertical tail is completely stalled so that the rudder effectiveness is essentially zero, and at $\beta = 15^\circ$ (at $V = 135$ knots), the rudder functions properly. A linear variation of $C_{n(\delta_r)}$ is assumed between $\beta = 15^\circ$ and $\beta = 20^\circ$.

(5) Total Control (reaction plus aerodynamic)

Again it is seen that sufficient control is provided for trim and maneuvering, the margins being in excess of the AGARD 408 requirements.

The control required levels for the N-309 correspond to the maximum VTO weight of 18,000 pounds.

Figures 2-38 and 2-39 Mod. T-39A Longitudinal and Lateral Trim (normal operation)

These figures are similar to Figures 2-34 and 2-36, and again demonstrate that ample control is available for trim and maneuvering.

The control required levels for the Mod. T-39A correspond to the maximum VTO weight of 20,200 lbs.

One further design requirement is with regard to lateral trim during engine run up; specifically, that in a 35 knot crosswind at T/W=0.9, roll moment required to trim must not exceed 50 percent of the "nominal" control moment (as specified in Section 2.4.1). This requirement is readily met on both N-309 and Mod. T-39A. The reason is that the control systems are designed to meet the design specifications at both maximum and minimum weights - including the control margin as discussed in (2)(a) above. Since at minimum weight the thrust to weight ratios for the N-309 and the Mod. T-39A in the composite mode are .76 and .79 respectively, it follows that the required margins are available at T/W=.90.

2.4.4 Stability Derivatives

Longitudinal and lateral-directional stability derivatives were determined for the N-309 in hovering, transition, and conventional flight modes. All cases considered were for the composite configuration with flaps and gear up. The assumption was made that the aircraft is a rigid body.

Table 2-4 shows a typical variation of the important derivatives with speed. It should be noted that the derivatives in the transition speed range ($V <$ 180 knots) are for level flight at a constant angle of attack of 7^{o}. This is significant since it means that in order to satisfy these conditions, the lift engines' thrust decreases with increasing speed. This and related points are further discussed later.

The sources of information used were the USAF Datcom and the Royal Aero. Society Data Sheets for the aerodynamic contributions, while the jet induced effects were based on the data on that subject included in this report. The approach used for each of the three flight modes is discussed below.

(a) Hover - It was assumed that any forces on the aerodynamic surfaces were negligibly small. Thus the hover derivatives consist wholly of engine inlet and exhaust effects, namely inlet momentum drag and jet induced local pressure changes on the aircraft structure.

The inlet momentum effects are a function of engine air mass flow and the local velocity of the inlet under consideration. For example, for a steady rate of pitch, the linear velocity of each pair of lift engines is different, while for a steady forward translation the incremental velocity is the same for all engines.

The flight condition analyzed was for $L = W = 18,000$ lb. at sea level.

(b) Conventional - Greatest emphasis was naturally placed on the low speed derivatives. Both longitudinal and lateral-directional derivatives were calculated at three angle of attack (0, 7^{o}, 14^{o}) for M = .25, W = 18,000 lb., aft c.g. and at the angle of attack for level flight at M = .75, W = 18,000 lb., h = 25,000 ft., aft c.g. In addition, the derivatives necessary to determine neutral point, maneuver point, and tail angle per g were calculated at intermediate Mach numbers.

Figures 2-40 and 2-41 show the variation of pitching moment and static directional stability (C_{n_β}) respectively with angle of attack at M = .25. It was assumed that $\alpha_o = 0$, $C_{m_o} = 0$.

(c) Transition - The transition derivatives are essentially the sum of the contributions of (a) and (b) above. Longitudinal and lateral-directional derivatives were estimated for a range of speeds (50, 100, 150 knots) at three angles

TABLE 2-4. CLEAN CONFIGURATION

L = W = 18,000 LB.

V ~ Knots EAS	0	50	100	150	180	M = .75
α ~ Deg		7	7	7	9	3
Altitude ~ Ft	0	0	0	0	0	25,000
X_u $\frac{1}{sec}$	-.0197	-.0235	-.0274	- .0295	- .00557	- .0236
Z_w $\frac{1}{sec}$	-.0197	-.2010	-.3770	- .5770	- .6890	- .8450
M_u $\frac{1}{sec\text{-}ft}$	.1353	.1709	.1920	.2053	0	0
M_w $\frac{1}{sec\text{-}ft}$	.000379	-.000429	-.000223	- .00500	- .01013	- .00834
M_q $\frac{1}{sec\text{-}rad}$	-.0078	-.1246	-.2413	- .3567	- .4062	- .5222
Y_v $\frac{1}{sec\text{-}rad}$	-.0197	-.0506	-.0812	- .1105	- .1048	- .1488
N_v $\frac{1}{sec\text{-}ft}$	-.000321	.00132	.00297	.00466	.00566	.01026
N_r $\frac{1}{sec\text{-}rad}$	-.00288	-.0800	-.1562	- .2335	- .2683	- .3721
L_v $\frac{1}{sec\text{-}ft}$	-.00952	-.01610	-.02175	- .02600	- .02395	- .03030
L_p $\frac{1}{sec\text{-}rad}$	-.00759	-.5403	-1.0665	-1.6017	-1.9102	-2.1586

STABILITY AND CONTROL - STABILITY DERIVATIVES - N-309

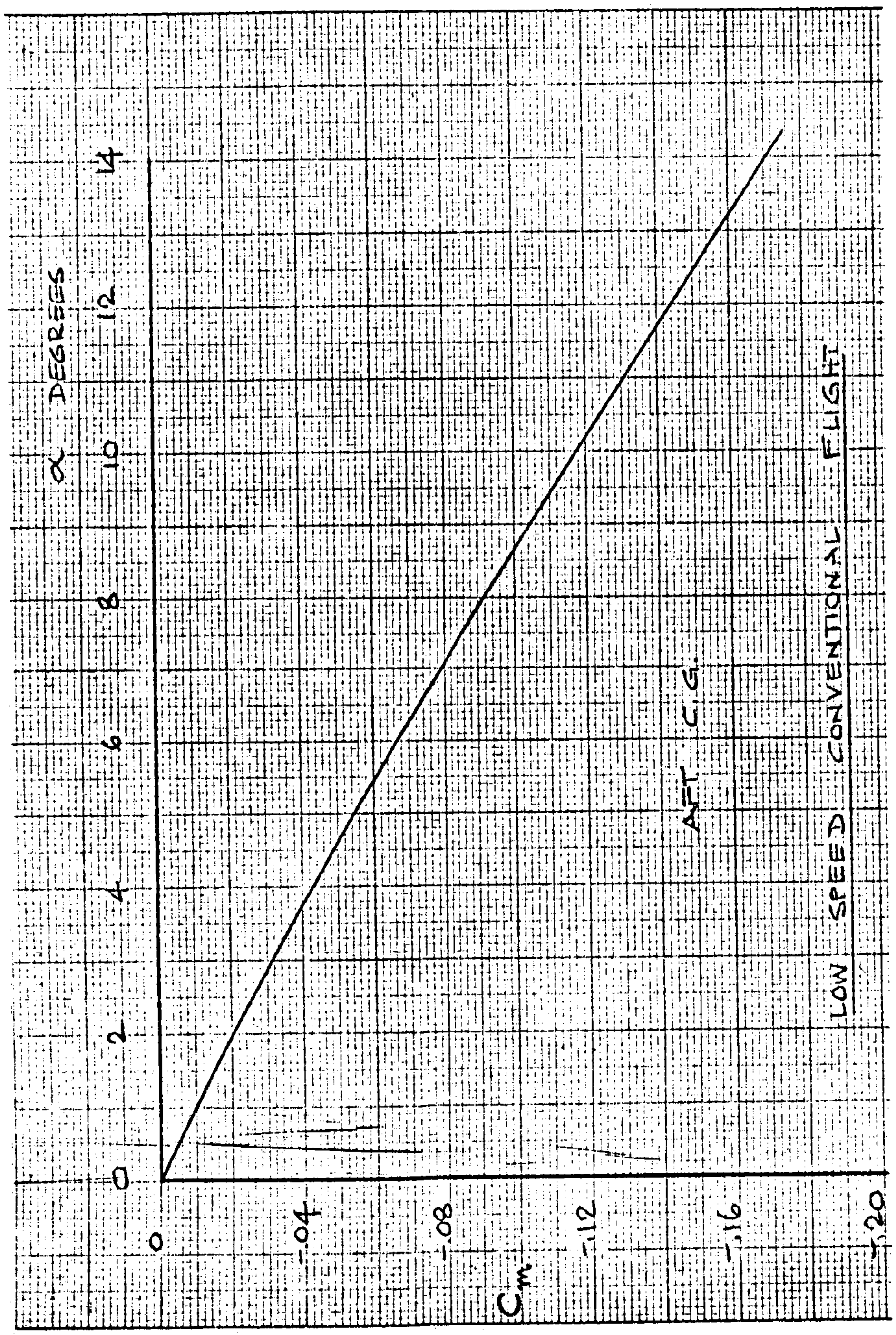


FIGURE 2-40. STABILITY AND CONTROL – STATIC LONGITUDINAL STABILITY – N-309

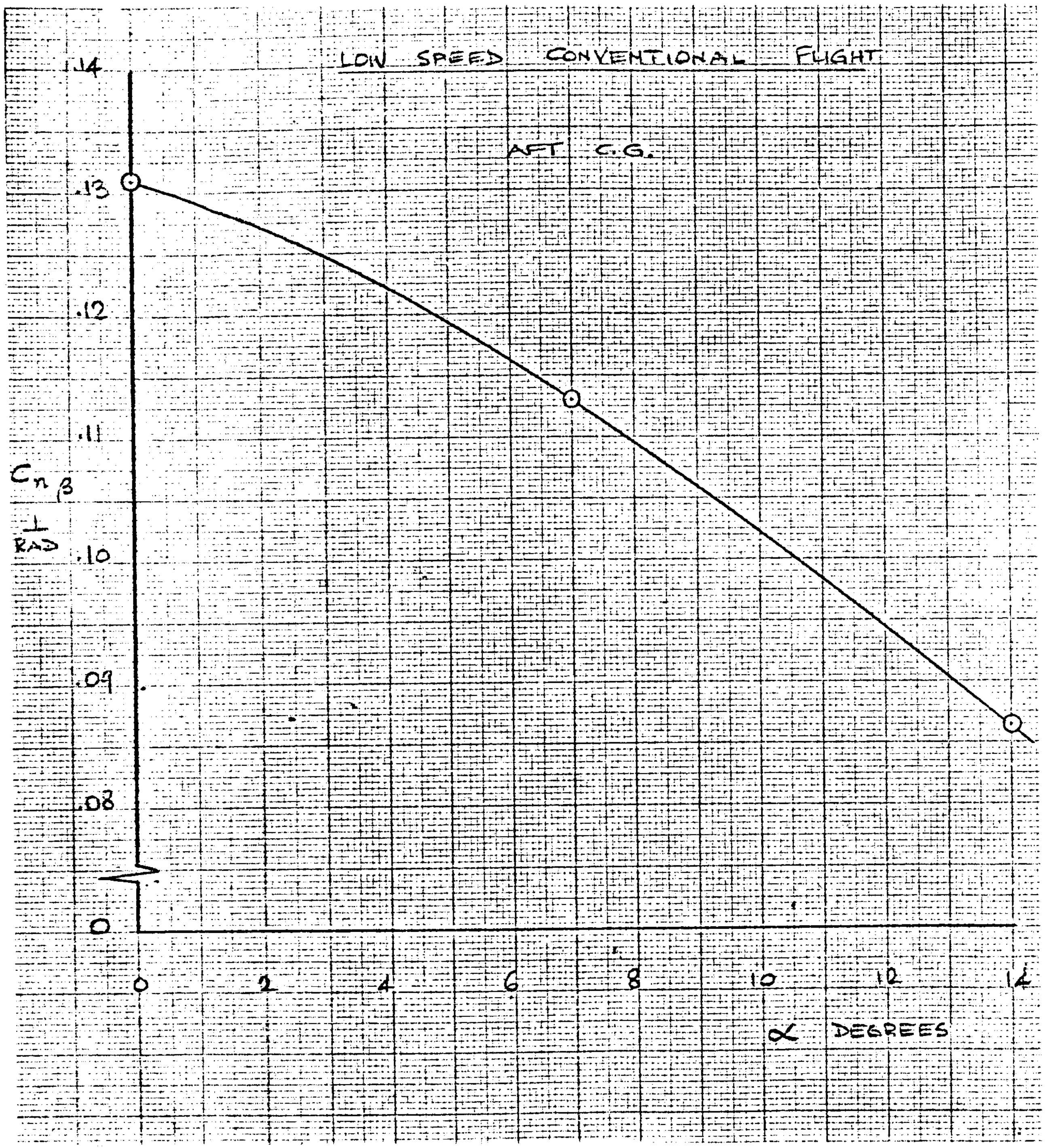


FIGURE 2-41. STABILITY AND CONTROL – STATIC DIRECTIONAL STABILITY VARIATION WITH ANGLE OF ATTACK – N-309

of attack (0, 7^o, 14^o) with the constraint that L = W = 18, 000 lb. This latter condition (L = W) is particularly significant in transition. It means that the throttle setting is different at each angle of attach (and indeed at each speed), since the lift engines provide a portion of the total lift. The point is that a number of the derivatives vary as a function of engine thrust or throttle setting. This comment applies to both the direct power effects (inlet and exhaust interference) and certain of the aerodynamic contributions. The former is in terms of air mass flow (inlet effects) and engine thrust (exhaust effects). The latter is the result of jet induced incremental downwash, and possibly sidewash, the magnitudes being a function of thrust level.

Sufficient data were available (Reference 2-23) to estimate an incremental downwash derivative $\Delta\frac{\partial \epsilon}{\partial \alpha}$ but no quantitative information could be found regarding the sidewash effect. It is known, however, from unpublished Hawker Aircraft (estimated) data that the P-1127 airplane shows a decrease in static directional stability and yaw damping as the engine nozzles are vectored aft. However, the P-1127 and the N-309 are so dissimilar that this information could not logically be applied; consequently the N-309 lateral-directional derivatives do not include any of these suspected effects.

The most important effect of the jet induced downwash is that it brings about a reduction in static longitudinal stability with respect to angle of attack (C_{m_α}) since $\frac{\partial \epsilon}{\partial \alpha}$ is increased. Secondary effects are seen in C_{L_α}. and $C_{m_{\dot{\alpha}}}$. As a result, the N-309 static stability is poorest at around 100 knots; this is shown in the data of Table 2-4. Again it is mentioned that these data are at a constant angle of attack of 7^o and the condition that L = W.

The derivatives, derived as described above, were used in the VSS, SAS and flying qualities studies.

2.4.5 Static Longitudinal Stability

Variation of the stick fixed neutral point and maneuver point with Mach number is shown in Figure 2-42 for the N-309.

It is seen from this plot that the minimum static margin at sea level is about 12 percent which, it could be argued, is too great. However, this preliminary analysis did not take into account any aeroelastic effects which generally reduce static stability

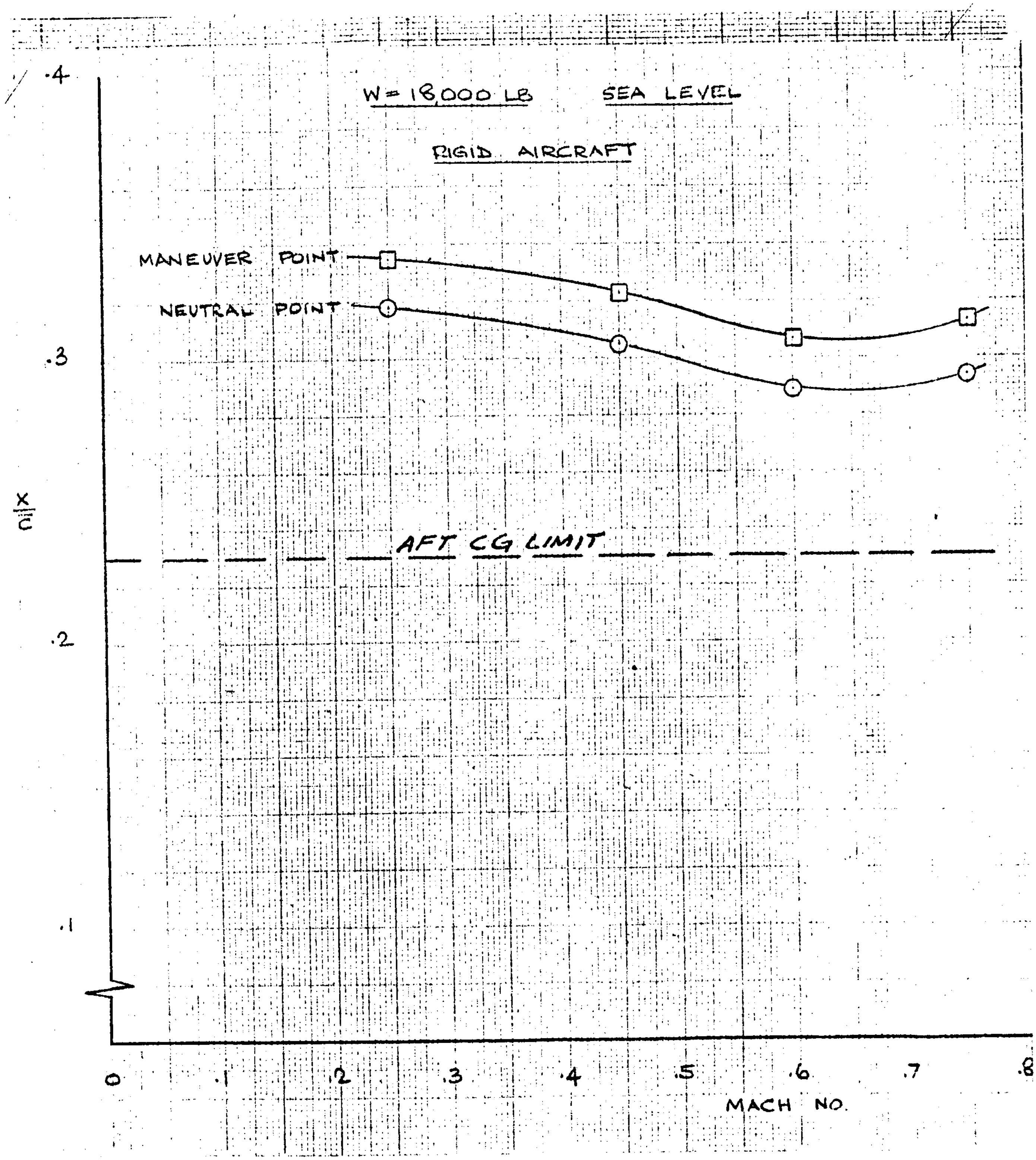


FIGURE 2-42. STABILITY AND CONTROL – STICK FIXED NEUTRAL POINT AND MANEUVER POINT – N-309

at the higher Mach numbers. It was therefore concluded that a horizontal tail area reduction was not justified based on the information currently at hand.

The variation of tail angle per g with Mach number is shown in Figure 2-43. Little probability of P.I.O. problems are indicated.

2.4.6 Longitudinal Dynamics

Short period mode characteristics were checked at two flight conditions, namely V = 180 knots at S.L. and M = .75 at 25,000 feet. In both cases, W = 18,000 lb. with the c.g. aft.

Figure 2-44 shows that with no stability augmentation the N-309 closely approaches the requirements of MIL-F-8785 (Amendment 4), paragraph 3.3.5. However, for absolute specification compliance, stability augmentation will be required.

At both flight conditions the Phugoid mode exhibits positive damping with a period in excess of 15 seconds (reference MIL-F-8785, paragraph 3.3.6).

2.4.7 Lateral-Directional Dynamics

Lateral-directional dynamics were analyzed at the same two conventional flight conditions as (2.4.6) above.

As shown in Figure 2-45, the N-309 Dutch Roll characteristics (with no stability augmentation) demonstrate compliance with the "augmenters on" requirement of MIL-F-8785, paragraph 3.4.1.

At V = 180 knots, S.L., the spiral mode doubles amplitude in 65 seconds, while at M = .75 at 25,000 feet, the spiral is convergent. Thus, compliance with MIL-F-8785, paragraph 3.4.2 requirement ($t_2 \nless 20$ sec.) is demonstrated.

Satisfactory lateral-directional characteristics are therefore indicated.

2.4.8 Roll Performance

A single degree of freedom analysis was made to determine the roll performance of the N-309 airplane in the conventional flight mode. The effects of yawing and sideslipping motions were neglected.

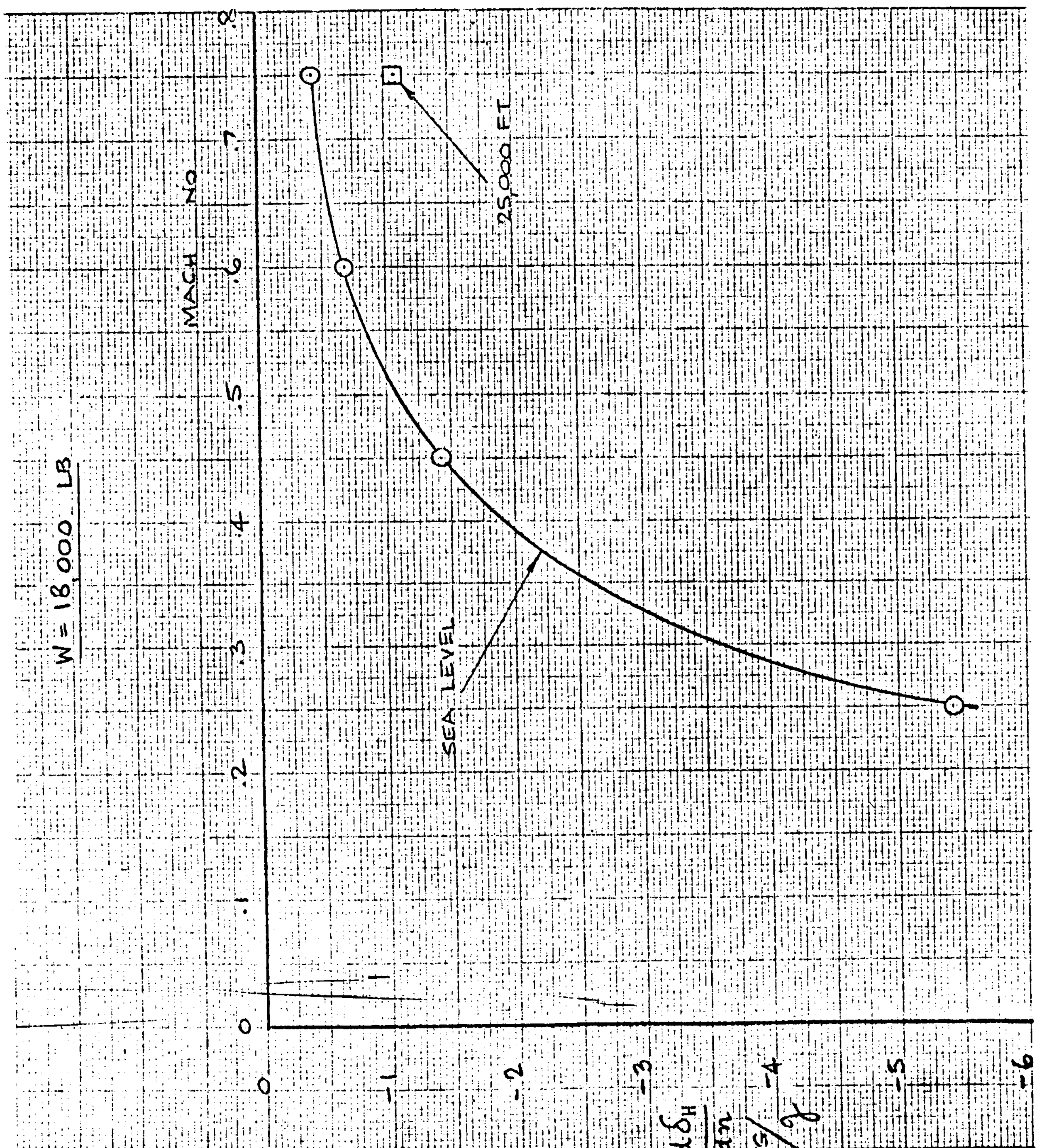


FIGURE 2-43. STABILITY AND CONTROL – TAIL ANGLE PER g – N-309

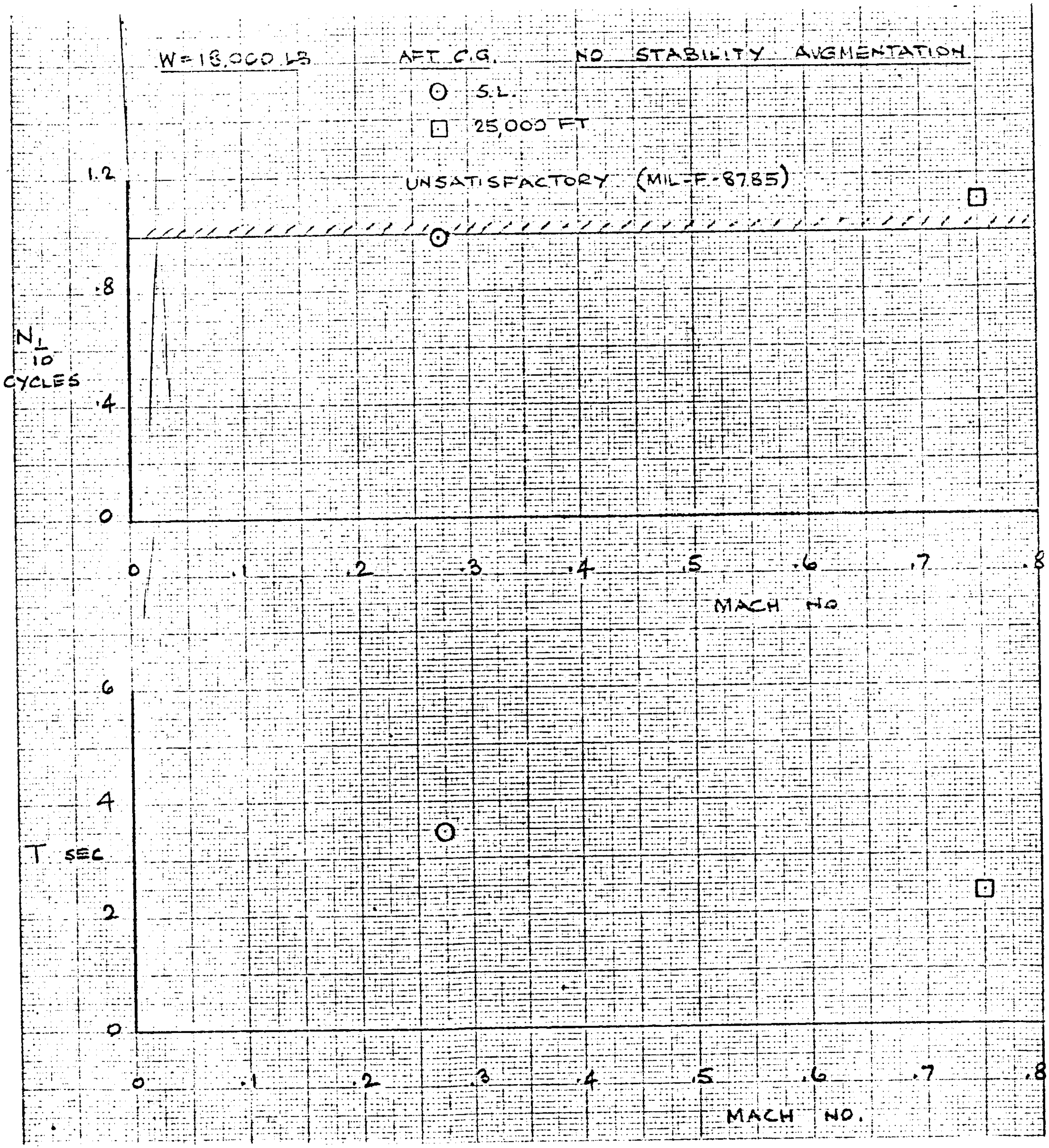


FIGURE 2-44. STABILITY AND CONTROL – LONGITUDINAL SHORT PERIOD CHARACTERISTICS IN CONVENTIONAL FLIGHT – N-309

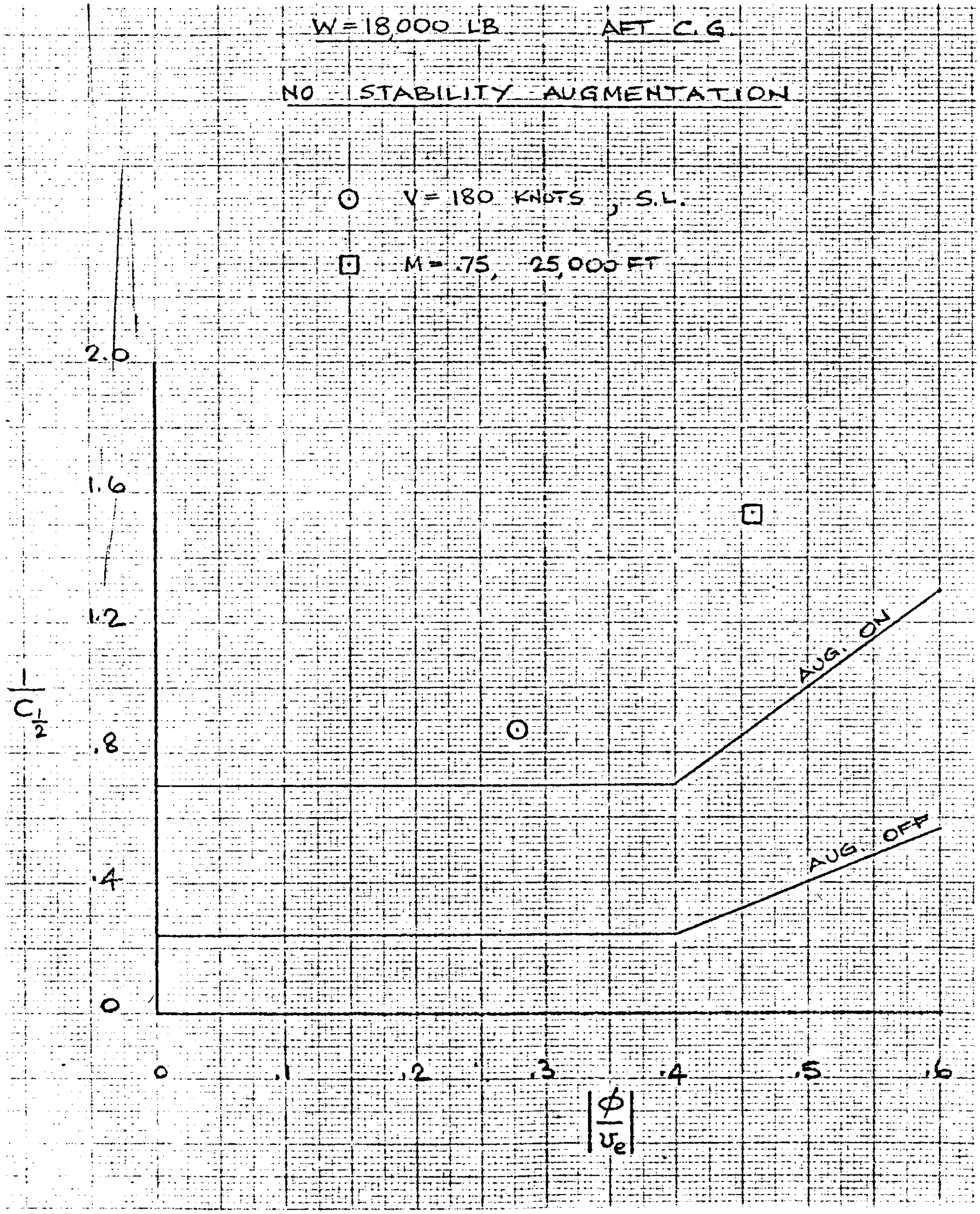


FIGURE 2-45. STABILITY AND CONTROL – DUTCH ROLL CHARACTERISTICS IN CONVENTIONAL FLIGHT – N-309

Using Table VI of Reference 2-16 as a guide, the following flight conditions were considered: W = 18,000 pounds at,

(a) h = S. L., $V = 1.1\ V_{stall}$ = 165 knots
(b) h = S. L., $V = V_{max}$ = 450 knots
(c) h = 25,000 feet, M = 0.75

Case (a) is somewhat academic since in general the lift engines will be operating at this low an airspeed so that the aircraft will not strictly be in conventional flight.

The pertinent roll performance data for a step input of full aileron ($\delta_a = 20^o$) are shown in Table 2-5. Considering the N-309 to be a Class III airplane, it is seen that the requirements of Reference 2-16 are satisfied.

TABLE 2-5. STABILITY AND CONTROL - ROLL PERFORMANCE - N-309

$\delta_a = 20^o$ W = 18,000 lb.

h ft.	S. L.	S. L.	25,000
M	0.25	0.68	0.75
V knots eas	165	450	302
$\frac{pb}{2V}$	0.1223 (0.07)	0.1223 (0.07)	0.1223
P_{ss} deg/sec	109.5	300	301
τ_R sec	0.570	0.208	0.463
ϕ deg t = 1	58.2	237	177 (90)

NOTES: (1) MIL-F-8785 (ASG) Amendment 4 requirements are shown in parenthesis.

(2) Also, at M = 0.25, average$\left(\frac{pb}{2V}\right)$ for first 30° of bank = 0.05 (0.05).

2.4.9 Gyroscopic Cross Coupling

Gyroscopic cross coupling exists on the N-309 and modified T-39A aircraft due to the angular momentum of the engines.

The lift/cruise engines, installed with their axes approximately parallel to the fuselage horizontal reference line, cause a pitching moment due to yawing velocity $\left(\frac{\partial M}{\partial r}\right)$ and a yawing moment due to pitching velocity $\left(\frac{\partial N}{\partial q}\right)$. The magnitudes of each term are equal and negligibly small.

The pure lift engines, installed with their axes generally perpendicular to the fuselage horizontal reference line, cause a pitching moment due to rolling velocity $\left(\frac{\partial M}{\partial p}\right)$ and a rolling moment due to pitching velocity $\left(\frac{\partial L}{\partial q}\right)$. The magnitudes of each are equal but of opposite sign. When taking the respective moments of inertia into consideration, it becomes apparent that the effect of $\frac{\partial M}{\partial p}$ is small, leaving only $\frac{\partial L}{\partial q}$ as a cross coupling term of any consequence. Under normal maneuvers (e.g., $q < 10$ deg/sec) this itself is also of little concern.

These comments may be verified by referring to the magnitudes of these terms which are listed in Table 2-6 for the N-309 and modified T-39A aircraft. The data are for composite mode operation with all engines at 100 percent rpm.

TABLE 2-6

STABILITY AND CONTROL - GYROSCOPIC CROSS COUPLING - N-309 AND MOD T-39A

Composite Mode Engines at 100% rpm

	N-309	MOD T-39A
$\frac{\partial M}{\partial r}$ ft. lb/deg/sec	29.9	29.9
$\frac{\partial M}{\partial p}$ ft. lb/deg/sec	104.7	119.7
$\frac{\partial L}{\partial q}$ ft. lb/deg/sec	-104.7	-119.7
$\frac{\partial N}{\partial q}$ ft. lb/deg/sec	29.9	29.9
W lb	18,000	20,200
I_x slug-ft^2	8,867	12,635
I_y slug-ft^2	45,482	44,436
I_z slug-ft^2	51,402	52,028

2.4.10 Modified T-39A - General

With the exception of hover-transition control power and gyroscopic effects, no stability and control analyses were performed for the Mod. T-39A. The assumption is that the modifications will not seriously degrade the already demonstrated acceptable flying qualities of the basic T-39A. Possible effects of the proposed modifications include:

1. While the increased fuselage volume is destabilizing, it is offset by the forward c.g. shift resulting from the redistribution of the fuselages' internal components. Thus, no revision to the empennage areas is considered necessary.
2. Increased moments of inertia will result in a somewhat less responsive airplane.

Since no stability and control data could be obtained for the basic T-39A, a "from scratch" analysis would have been required, which in view of the above comments was not considered warranted.

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2-23. Margason, R.J. and Gentry, G.L., LWP-325, Aerodynamic Characteristics of a Five-Jet VTOL Configuration in the Transition Speed Range, NASA Langley, November 1966.

NOMENCLATURE FOR TABLE 2-4

$X_u = \frac{1}{m}\frac{\partial X}{\partial u} = \frac{\rho SU}{m}(-C_D - C_{D_u})$	$\frac{1}{sec}$
$Z_w = \frac{1}{m}\frac{\partial Z}{\partial w} = \frac{\rho SU}{2m}(-C_{L_\alpha} - C_D)$	$\frac{1}{sec}$
$M_u = \frac{1}{I_y}\frac{\partial M}{\partial u} = \frac{\rho SU\bar{c}}{I_u}(C_m + C_{m_u})$	$\frac{1}{sec\text{-}ft}$
$M_w = \frac{1}{I_y}\frac{\partial M}{\partial w} = \frac{\rho SU\bar{c}}{2I_y} C_{m_\alpha}$	$\frac{1}{sec\text{-}ft}$
$M_q = \frac{1}{I_y}\frac{\partial M}{\partial q} = \frac{\rho SU\bar{c}^2}{4I_y} C_{m_q}$	$\frac{1}{sec\text{-}rad}$
$Y_v = \frac{1}{m}\frac{\partial Y}{\partial v} = \frac{\rho SU}{2m} C_{y_\beta}$	$\frac{1}{sec\text{-}rad}$
$N_v = \frac{1}{I_z}\frac{\partial N}{\partial v} = \frac{\rho SUb}{2I_z} C_{n_\beta}$	$\frac{1}{sec\text{-}ft}$
$N_r = \frac{1}{I_z}\frac{\partial N}{\partial r} = \frac{\rho SUb^2}{4I_z} C_{n_r}$	$\frac{1}{sec\text{-}rad}$
$L_v = \frac{1}{I_x}\frac{\partial L}{\partial v} = \frac{\rho SUb}{2I_x} C_{l_\beta}$	$\frac{1}{sec\text{-}ft}$
$L_p = \frac{1}{I_x}\frac{\partial L}{\partial p} = \frac{\rho SUb^2}{4I_x} C_{l_p}$	$\frac{1}{sec\text{-}rad}$

3.0 PROPULSION PERFORMANCE

3.1 ENGINE INSTALLATION

General Electric YJ85-19 engines were selected for lift and lift/cruise engines in both aircraft designs. A typical lift-engine bay is shown in Figure 3-1 (AD 4515). As indicated, there is one set of bomb-bay type inlet doors for the forward four lift engines; similar doors are installed for the aft engines. These doors are hinged so that they provide somewhat of a continuation of the inlet bell mouth and as deflectors to increase the reingestion flow path. The doors are electrically driven through flex shafting to jacks at each side of the airplane. A single jack screw for each pair of doors assures symmetric operation. Lift engine RPM switches prevent door closing at an engine speed greater than 12 percent RPM. The lift-engine exhaust doors are similarly arranged and powered. Suck-in inlets and pop-out exhaust doors insure that structural damage will not occur when the doors shut.

Spherical nozzles are provided to vector thrust of the lift engines. The lift/cruise engines are provided with thrust vectoring doors with an actuator for each door. The vector doors are electrically operated and linked electrically to the lift/cruise engine thrust diverter valve such that the doors must be open before the diverter valves can be opened. The vectoring doors open through 80 degrees of travel with the last 18 degrees controllable with the thrust vector control on the throttle knob.

Engine bay cooling air is drawn in around the outside of the inlet bell mouths and expelled by air ejectors installed in the engine bays.

3.2 BARE ENGINE PERFORMANCE

The YJ85-19 lift engine has a bare standard day take-off thrust rating of 3,015 pounds and an SFC of 1.00. The operational time limit is 5.0 minutes at take-off power. These ratings apply to the lift/cruise engine in the lift mode for composite (mixed engines) hover operation. For cruise operation, bare-engine ratings are 2,950 pounds for thrust and an SFC of 0.98. The reduction in thrust rating in the cruise model is due to the lowered allowable exhaust gas temperature for extended 30-minute military power operation. Time between overhauls is 400 hours, which is sufficient for the test programs without incurring a cost penalty for engine overhauls.

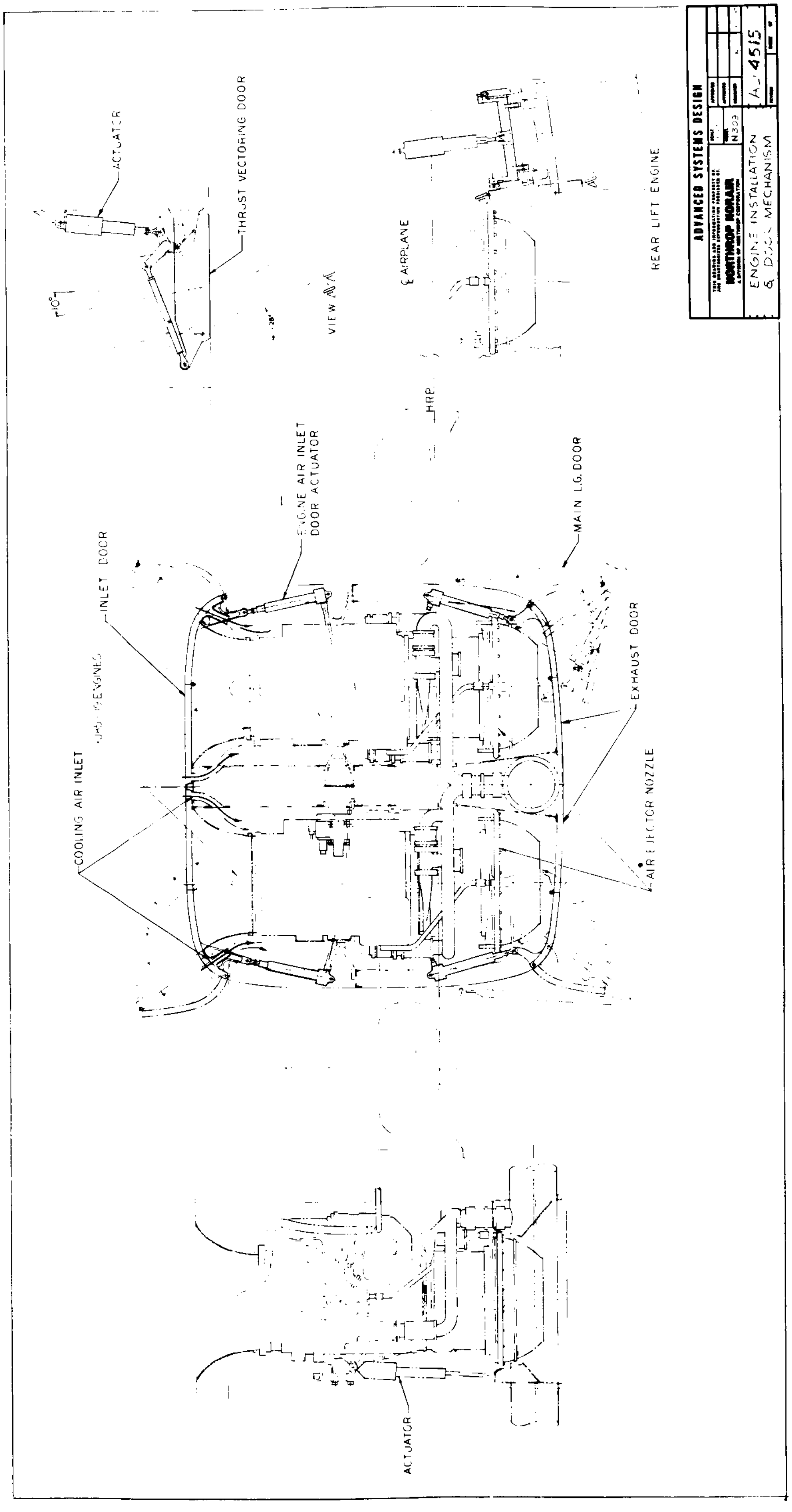

ACTUATOR
THRUST VECTORING DOOR
10°
VIEW A-A
℄ AIRPLANE
REAR LIFT ENGINE
HRP
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
ENGINE INSTALLATION
A-4515
ENGINE AIR INLET DOOR ACTUATOR
MAIN L.G. DOOR
INLET DOOR
COOLING AIR INLET
EXHAUST DOOR
AIR EJECTOR NOZZLE
ACTUATOR

FIGURE 3-1

3.3 ENGINE BLEED OPERATION

The operational requirements of the NASA V/STOL designs include hover testing for extended periods with high compressor bleed rates for attitude control. Therefore, exhaust tailpipes were sized to allow continuous engine operation at maximum power and compressor bleed rate without exceeding the limiting 5-minute exhaust gas temperature of 1780°R (computed).

The effect on engine performance of resizing the exhaust area to retain the limiting EGT while bleeding 8.0 and 10.0 percent is shown in Figure 3-2. At maximum power and the maximum bleed rate of 10.0 percent, the engine thrust loss is 16 percent with a 9.0 percent increase in engine specific fuel consumption. However, where, as in the proposed designs, the bleed air is utilized for lift, the actual lift loss is reduced to about 7 percent with essentially no penalty on SFC. This comparison assumes the complete utilization of bleed air for lift with no losses in the control system.

The effect of off-design bleed rates on engine performance and lift capability is shown in Figure 3-3 for an engine sized for 10 percent bleed operation. The important points are: (1) lift capability (engine thrust plus control thrust) is only slightly reduced for bleed rates 2 to 3.0 percent less than design; and (2) specific fuel consumption based on total lift decreases for bleed rates less than design.

A rapid reduction of bleed rate results in a loss of engine stall margin due to surge effects on the compressor. Engine tolerance to bleed-rate changes depends on the particular design, varying from a permissible 0.5 percent reduction from the design bleed rate for a current foreign lift engine of high thrust-to-weight rate, to no restriction for the YJ85-19 engine, a slightly modified conventional turbojet engine.

As discussed in detail under a separate heading, the control systems of the new and modified aircraft utilize the variable bleed concept with bleed rates varying between about 8.0 and 10.0 percent, depending on the control demand.

3.4 INSTALLED PERFORMANCE

3.4.1 General

The primary NASA test condition is at sea level and an ambient temperature of 80°F. Performances of the engines at this condition at maximum power as installed in the new design (N-309) and the modified T-39A are listed in Figure 3-4. Also

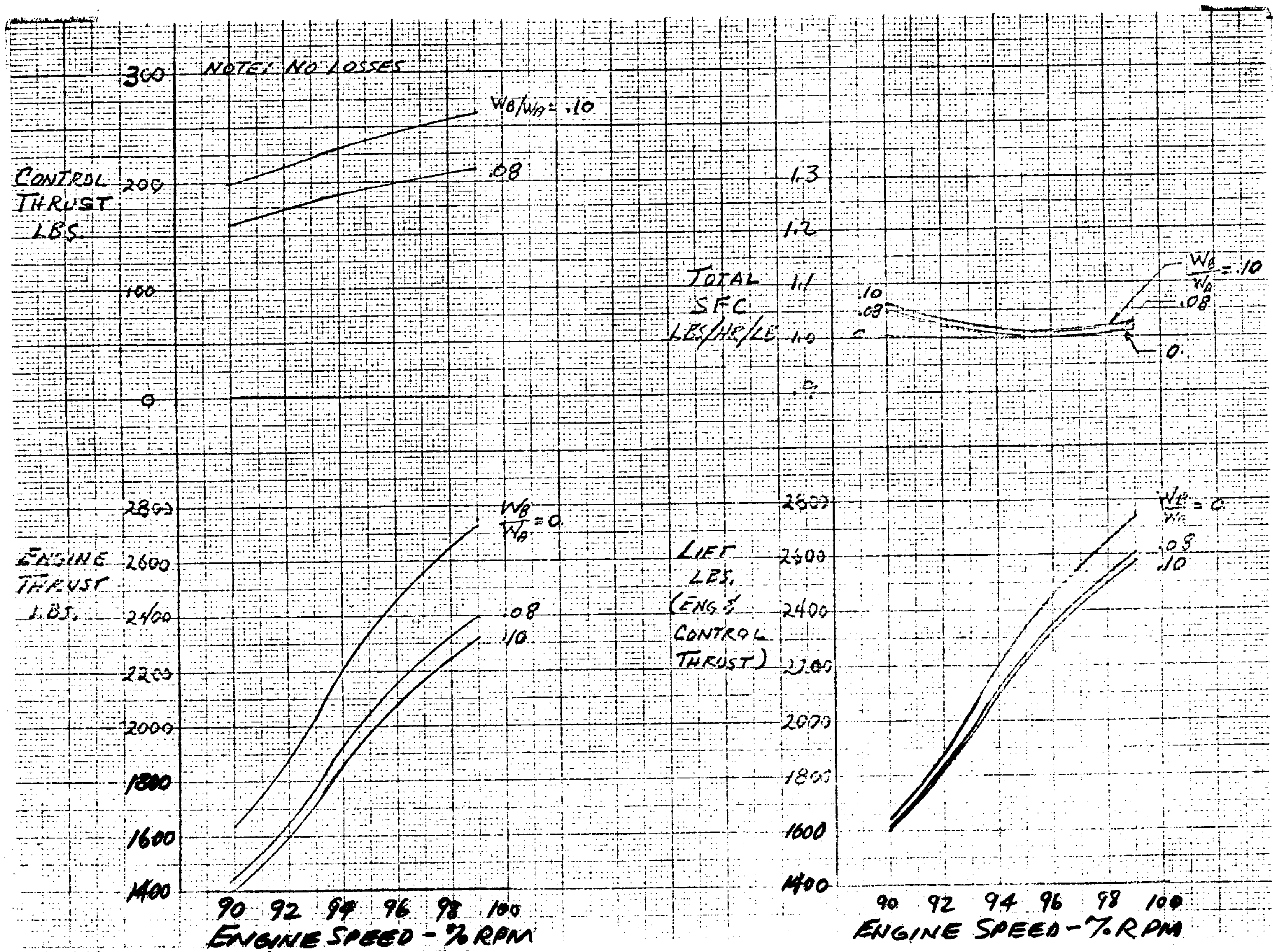


FIGURE 3-2. EFFECT OF RESIZING EXHAUST NOZZLE FOR MAXIMUM POWER CONSTANT BLEED OPERATION INSTALLED YJ85-19 ENGINE SEA LEVEL 80°F

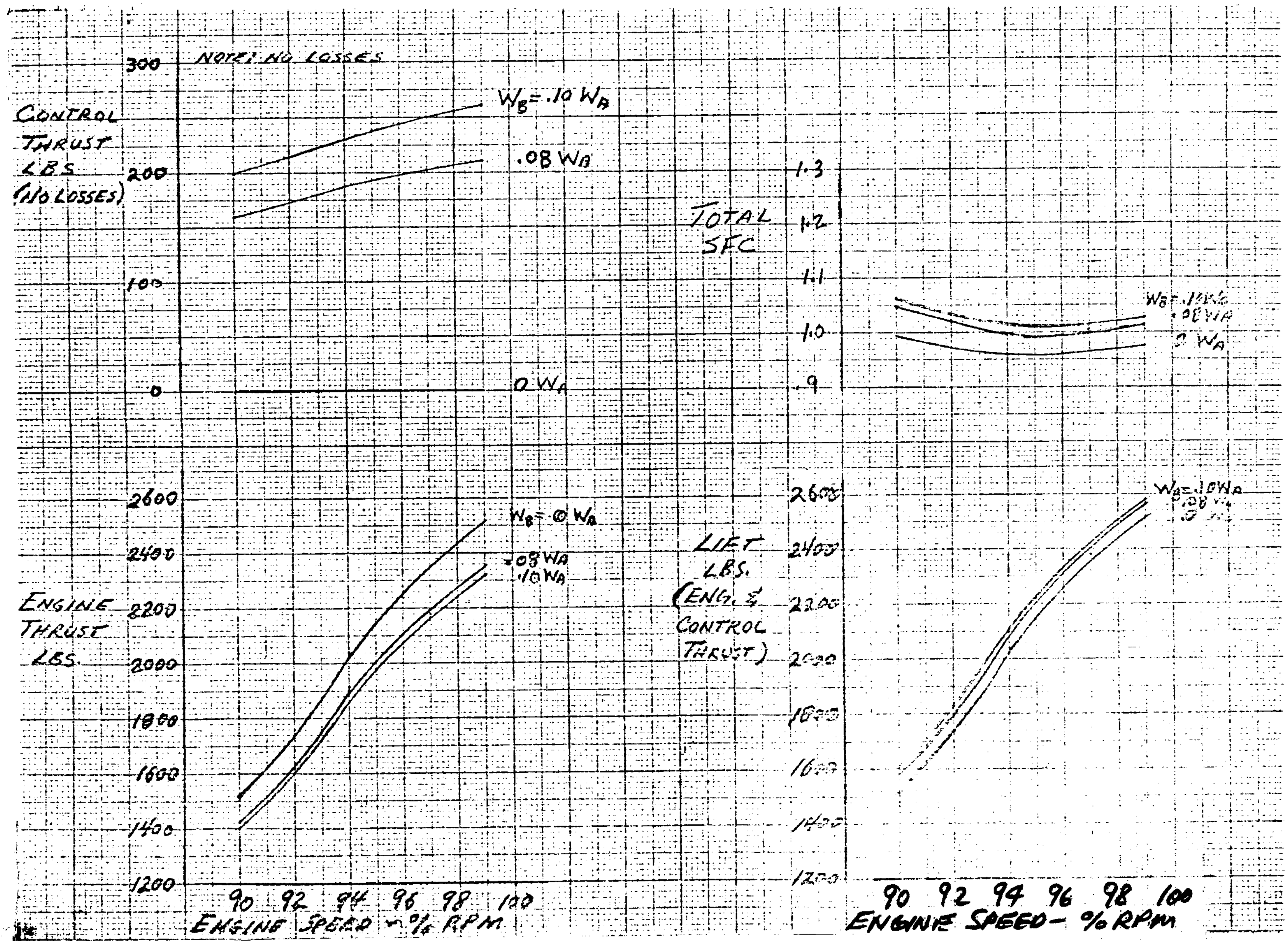


FIGURE 3-3. EFFECT OF BLEED RATE ON ENGINE PERFORMANCE EXHAUST NOZZLE SIZED FOR CONSTANT 10 PERCENT BLEED RATE INSTALLED YJ85-19 ENGINE SEA LEVEL, 80°F DAY

SUMMARY OF YJ85-19 ENGINE PERFORMANCE

(Installed Static Sea-Level Ratings at 80°F and Maximum Bleed Rate, $W_B/W_A = 0.10$)

	New Aircraft N-309 Lift	New Aircraft N-309 L/C Lift Mode	Modified T-39A Lift	Modified T-39A L/C Lift Mode	New And Mod. A/C T-39 (4) Cruise
(1) Engine Weight lb	420	392	420	392	392
(2) Thrust lb	2320	2250	2320	2260	2680
(3) Control Thrust, F_c lb	226	211	226	211	-
Thrust/Weight (Engine only)	5.52	5.74	5.52	5.75	6.84
Thrust/Weight(Eng. plus F_c)	6.06	6.27	6.06	6.27	-
Engine SFC lb/lb-hr	1.11	1.16	1.11	1.16	1.038
Total SFC lb/lb-hr	1.012	1.06	1.012	1.06	-
Bleed Press. at Port Exit PSIA	78.5	78.2	78.5	78.2	-
Bleed Temp. at Port Exit °R	965	965	965	965	-
Comp. Bleed Air Rate lb/sec	4.18	4.16	4.18	4.16	-
Control Nozzle Specific Thrust F_c/W_B	56.9	56.4	56.9	56.4	-

(1) Includes vectoring nozzle for lift engine but not diverter valve and extended tailpipe for L/C engine

(3) Control System Losses
- A. Line press. loss, $\Delta p/p = 0.15$
- B. Bleed Air noz. leakage, .03 W_B
- C. Nozzle velocity coeff., 0.96
- D. Bay cooling, $W_c = .2$ lb/sec
- E. Air cond. (L/C only), W = .2 lb/sec

(4) Bleed Rate, $W_B/W_A = 0.01$

(2) Installation Losses, $\Delta F/F$

Lift Engine
- A. .007 (.995 inlet recv.)
- B. .015 (Vect. nozzle)

L/C Engine
- A. .014 (.99 inlet recv.)
- B. .033 (Div. and tailpipe, new A/C)
- C. .028 (Div. Mod. T-39)

Cruise Mode
- A. .014 (.99 inlet recv.)
- B. .025 (Div. and tailpipe)

FIGURE 3-4. SUMMARY OF YJ85-19 ENGINE PERFORMANCE listed are other pertinent performance parameters, including engine weights, bleed air quality, and available control thrust at the maximum compressor bleed rate of 10.0 percent. These data reflect the tabulated installation losses which are discussed below, as are the overall effects of resizing the exhaust tailpipes for continuous bleed operation.

3.4.2 Lift Engine

Static performance of the YJ85-19 lift engine at sea level and 80^{o}F as installed in either the new (N-309) or modified aircraft is shown in Figure 3-5 for compressor bleed rates varying from 6 to 10 percent. An increase of nozzle area from 104.1 square-inches for the basic engine to 109.5 square inches (effective) was required to retain the limiting exhaust gas temperature of 1780^{o}R when bleeding 10.0 percent.

Installation thrust losses consisted of 0.7 percent for an inlet recovery of 0.995 and 1.5 percent for the vectoring exhaust nozzle in the undeflected position. These losses yielded an installed thrust rating of 2,320 pounds and an engine specific fuel consumption of 1.11 at a bleed rate of 10.0 percent.

3.4.3 Lift/Cruise Engine

Installed lift performance of the lift/cruise engine is shown in Figure 3-6. These data are applicable to both aircraft although, as shown in Figure 3-4, the MOD. T-39A lift/cruise engine is rated at 10 pounds more thrust, 2,260 pounds against 2,250 pounds for the N-309. This small difference was due to a slightly less pressure drop in the T-39A diverter-tailpipe, as shown in Figure 3-7. This figure presents tailpipe pressure losses for each configuration and mode of operation used in the performance calculations. Diverter gas leakage rates are 0.90 and 0.70 percent of the total gas flow in the lift and cruise modes, respectively.

Lift thrust and fuel flow for the lift/cruise engine up to 0.30 Mach number on a sea-level standard day and 80^{o}F are shown in Figures 3-8 and 3-9, respectively.

Maximum static cruise thrust at sea level and 80^{o}F is 2,680 pounds for both installations, i.e., diverter valve in the straight through position and a 20.0 inch tailpipe and nozzle. In the cruise mode, the engine was sized for constant military operation at 80^{o}F, a bleed rate of 1.0 percent for bay cooling and cockpit conditioning, and a power extraction of 30 horsepower.

Cruise performance, thrust and fuel flow, up to 0.30 Mach number on a sea level, 80^{o}F day are shown in Figures 3-10 and 3-11. Similar standard day

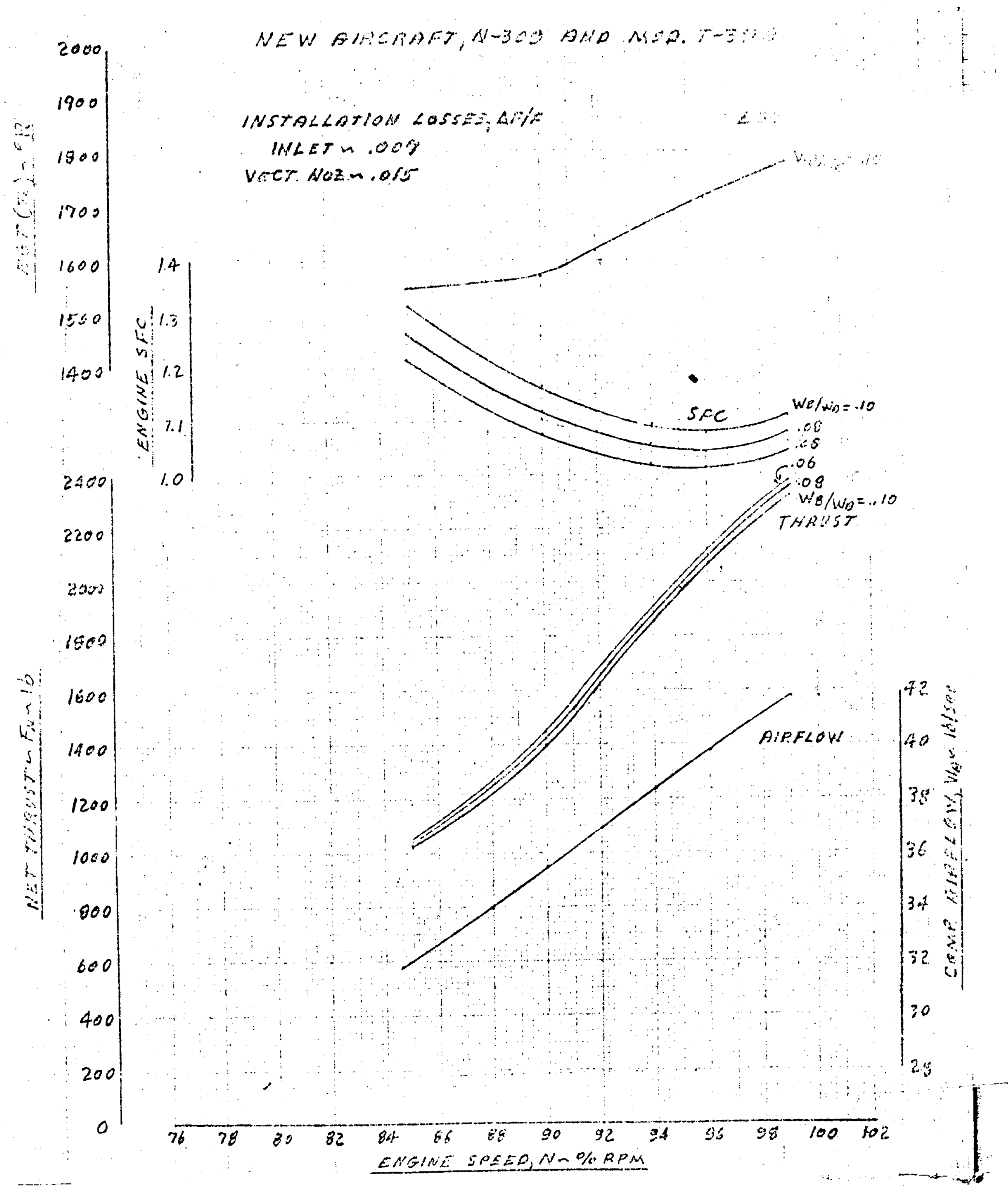


FIGURE 3-5. YJ85-19 LIFT ENGINE PERFORMANCE INSTALLED, SEA-LEVEL, 80°F EXHAUST NOZZLE SIZED FOR CONSTANT 10 PERCENT BLEED

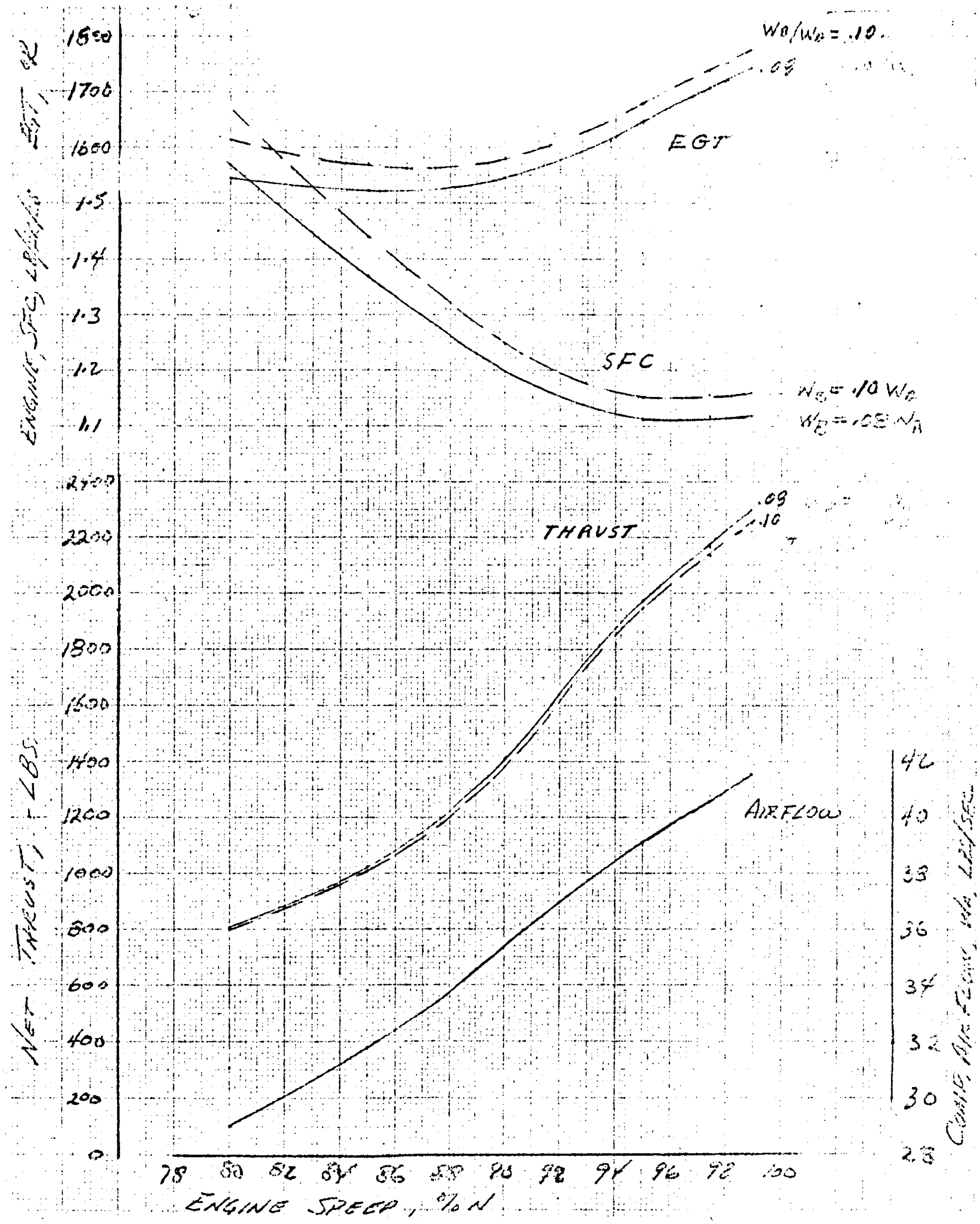


FIGURE 3-6. YJ85-19 LIFT/CRUISE ENGINE PERFORMANCE IN LIFT MODE DIVERTER VALVE PLUS 60° BEND INSTALLED, SEA LEVEL, 80°F, HP = 30 EXHAUST NOZZLE SIZED FOR CONSTANT 10 PERCENT BLEED (A_B = 114.46 IN^2)

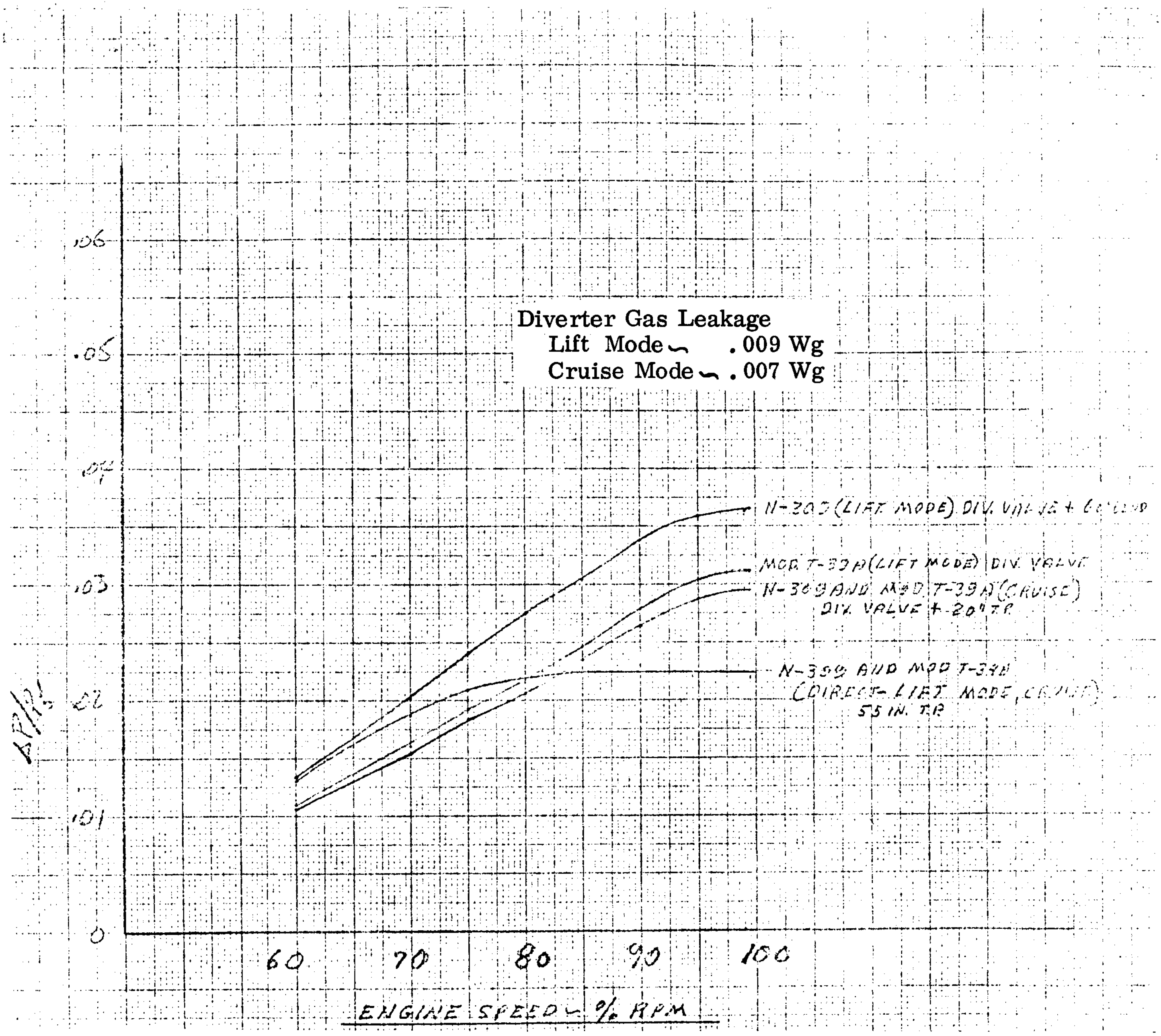


FIGURE 3-7. YJ85-19 LIFT/CRUISE ENGINE DIVERTER VALVE AND TAILPIPE PRESSURE DROP VERSUS ENGINE SPEED

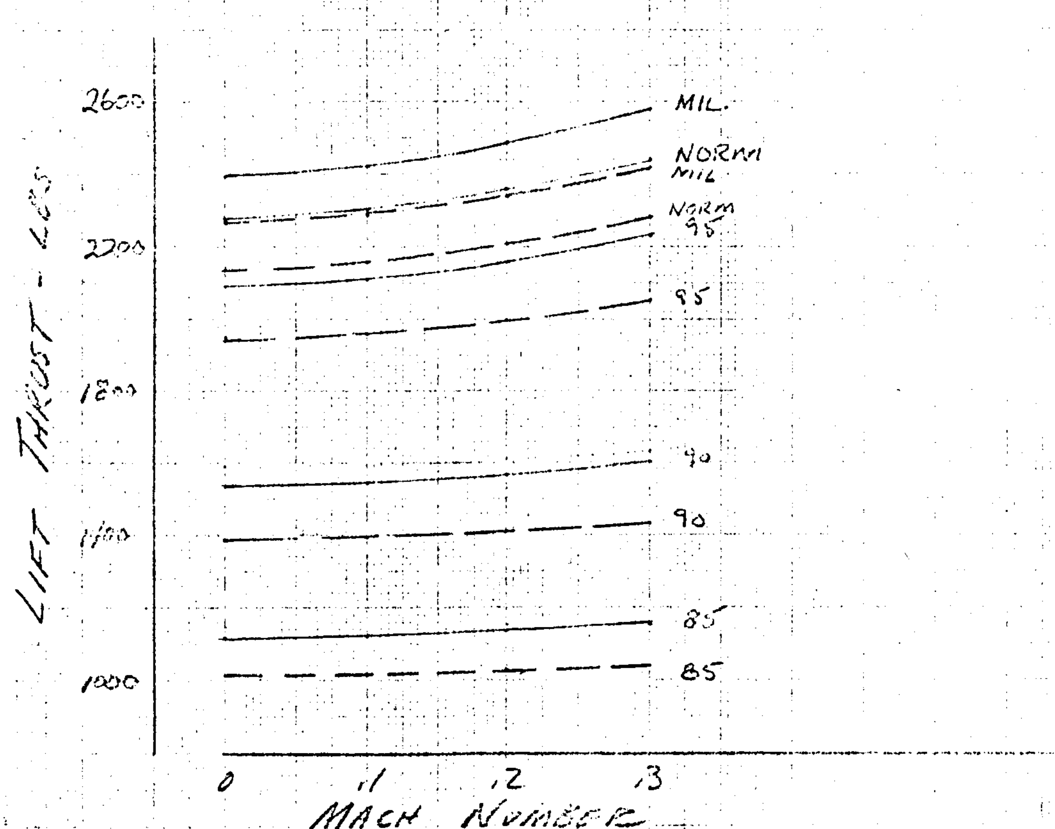


FIGURE 3-8. LIFT THRUST VERSUS MACH NUMBER AND POWER SETTING - SEA LEVEL DIVERTER VALVE

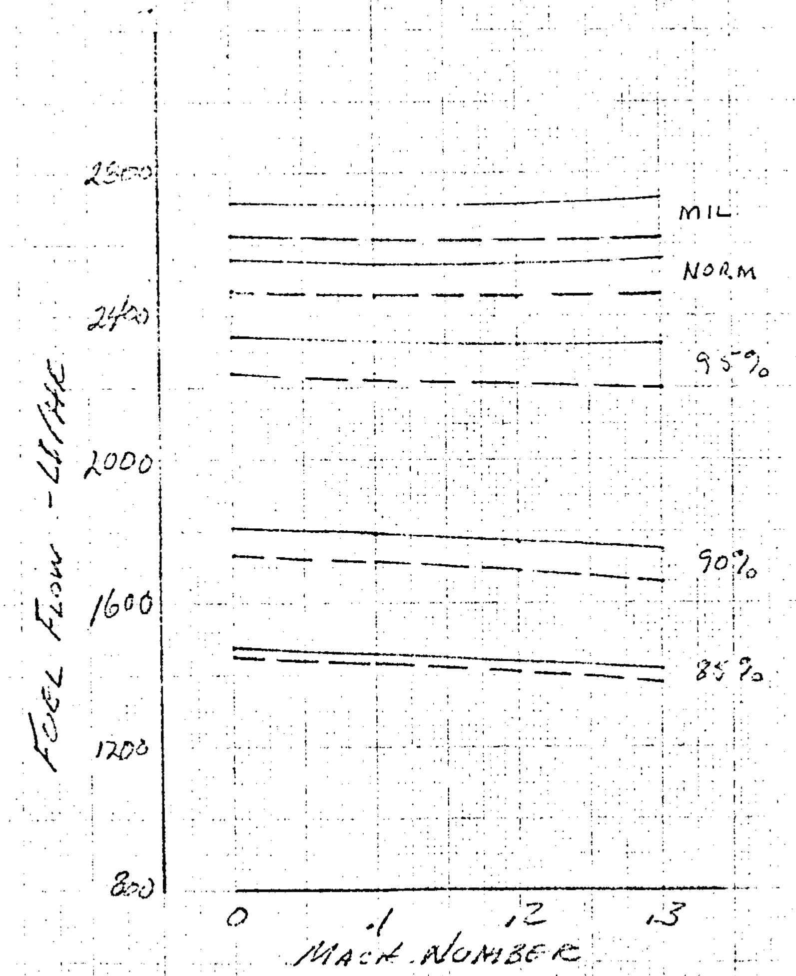


FIGURE 3-9. LIFT FUEL FLOW VERSUS MACH NUMBER AND POWER SETTING - SEA LEVEL

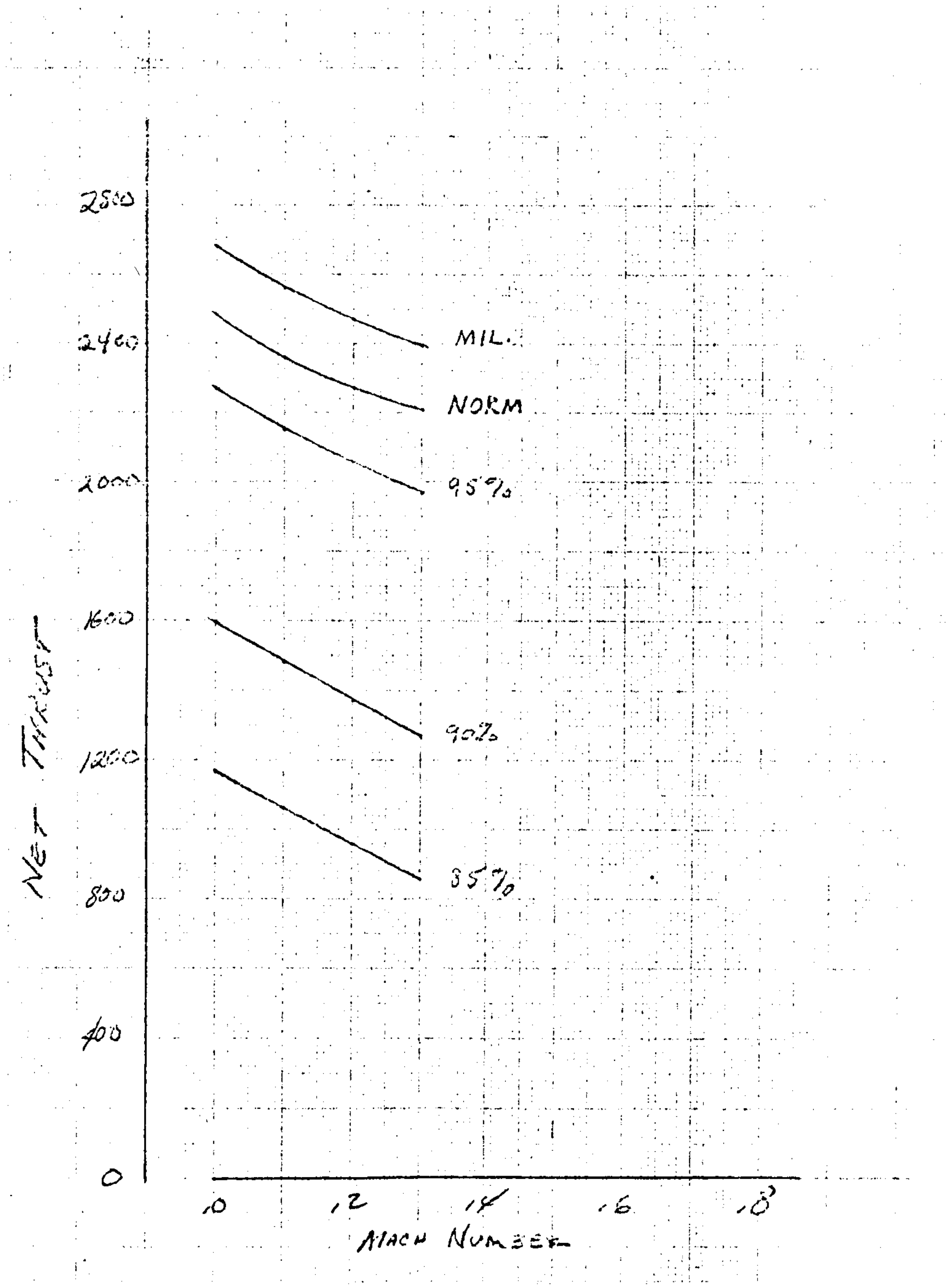


FIGURE 3-10. CRUISE THRUST VERSUS MACH NUMBER AND POWER SETTING - SEA LEVEL 80°F DIVERTER VALVE +20 INCH TAILPIPE

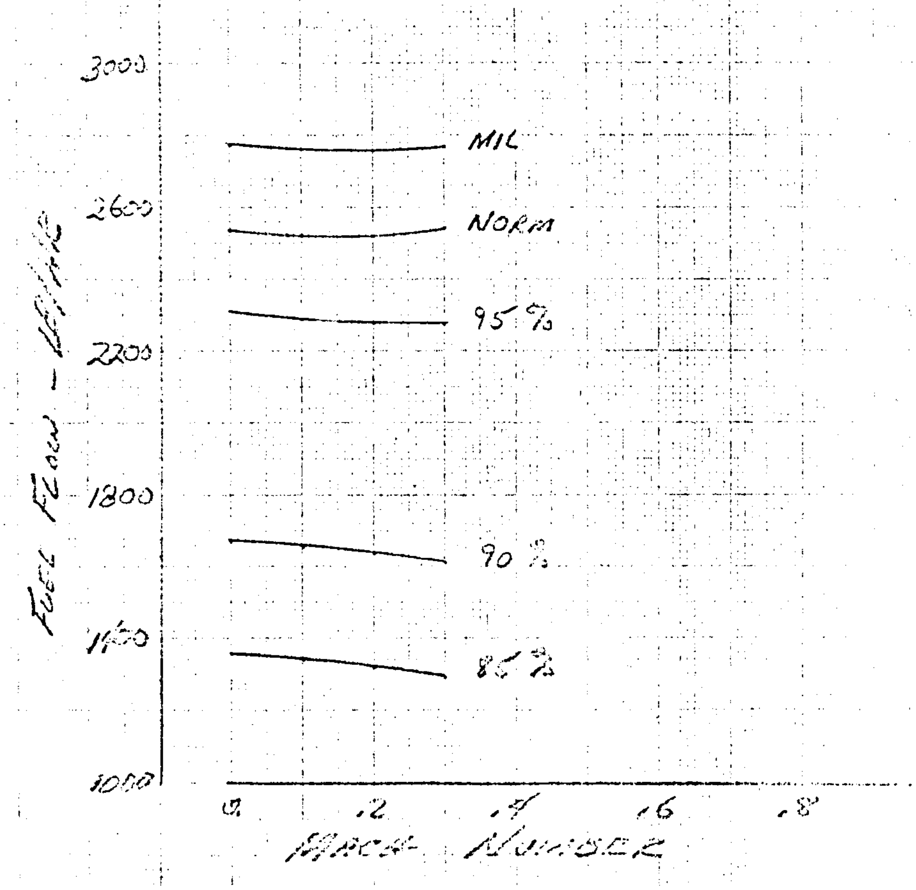


FIGURE 3-11. CRUISE FUEL FLOW VERSUS MACH NUMBER AND POWER SETTING - SEA LEVEL 80°F DIVERTER VALVE +20 INCH TAILPIPE

performances at sea level and altitudes of 15,000 and 25,000 feet are presented in Figures 3-12 through 3-17.

3.5 LIFT CAPABILITY

Maximum lift capability of each airplane in composite-and direct-lift modes of operation are shown in Figures 3-18 through 3-21. These data include the entire control thrust as lift. In the actual case with yaw and roll moments applied, lift is reduced somewhat due to the swivel angle of the pitch nozzle for yaw and the upward exhaust of one of the roll nozzles at high roll moments.

3.6 ENGINE BAY COOLING

Cooling of the lift and lift/cruise engine bays is accomplished by jet pumps using a maximum of 0.5 percent compressor bleed air from each engine. The ejector consists of a circular tube located just above the spherical nozzle with nozzles spaced equally around the periphery. The system is sized to provide a secondary airflow of about 2.0 to 2.5 lb/sec. This required a secondary-to-primary area ratio of about 350. The primary system consists of twenty 0.11 inch diameter nozzles with the engine bay in the same horizontal plane as the nozzles modified to provide the required area ratio.

The secondary flow path is through cutouts in the upper fuselage outside the bell mouth proper of the lift engines, around the compressor for cooling engine accessories, then through the ejector. The total area of the inlet cutouts is 38.0 square-inches to limit the entry Mach number to about 0.13, thereby holding flow pressure losses to a minimum.

For the lift/cruise engine, the ejector is mounted near the exit of the cruise tailpipe. The ejector is in operation at all times to provide cooling air flow during both the lift and cruise modes of operation.

3.7 INLET PERFORMANCE

3.7.1 Lift Engine

Design objectives of the lift-engine inlet were: (1) provide a high recovery during hover so as to not compromise lift; (2) provide recoveries and compressor face distortions compatible with the engine during start, acceleration and high power

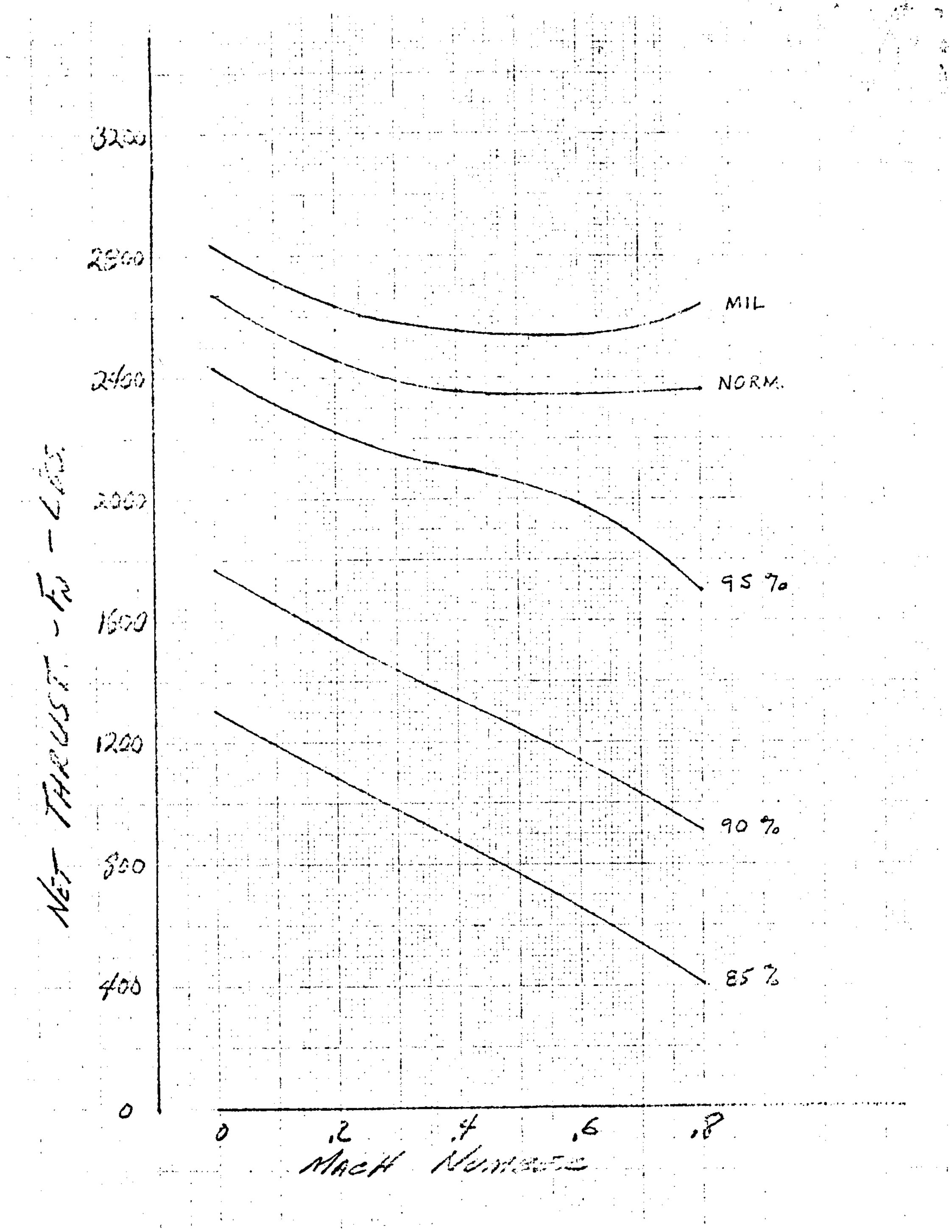


FIGURE 3-12. CRUISE THRUST VERSUS MACH NUMBER AND POWER SETTING FOR SEA LEVEL STANDARD DAY DIVERTER VALVE +20 INCH TAILPIPE

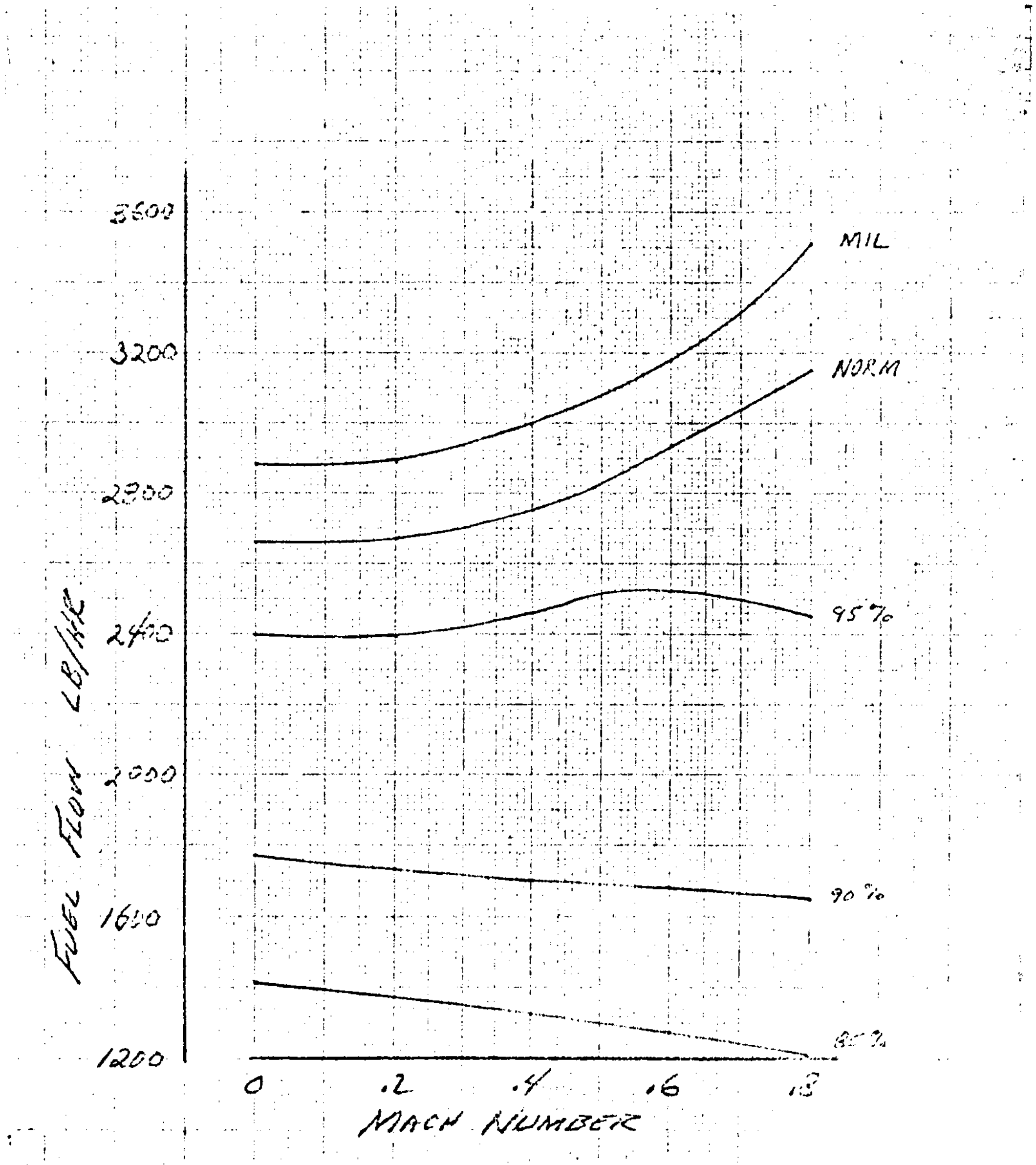


FIGURE 3-13. CRUISE FUEL FLOW VERSUS MACH NUMBER AND POWER SETTING – SEA LEVEL STANDARD DAY DIVERTER VALVE +20 INCH TAILPIPE

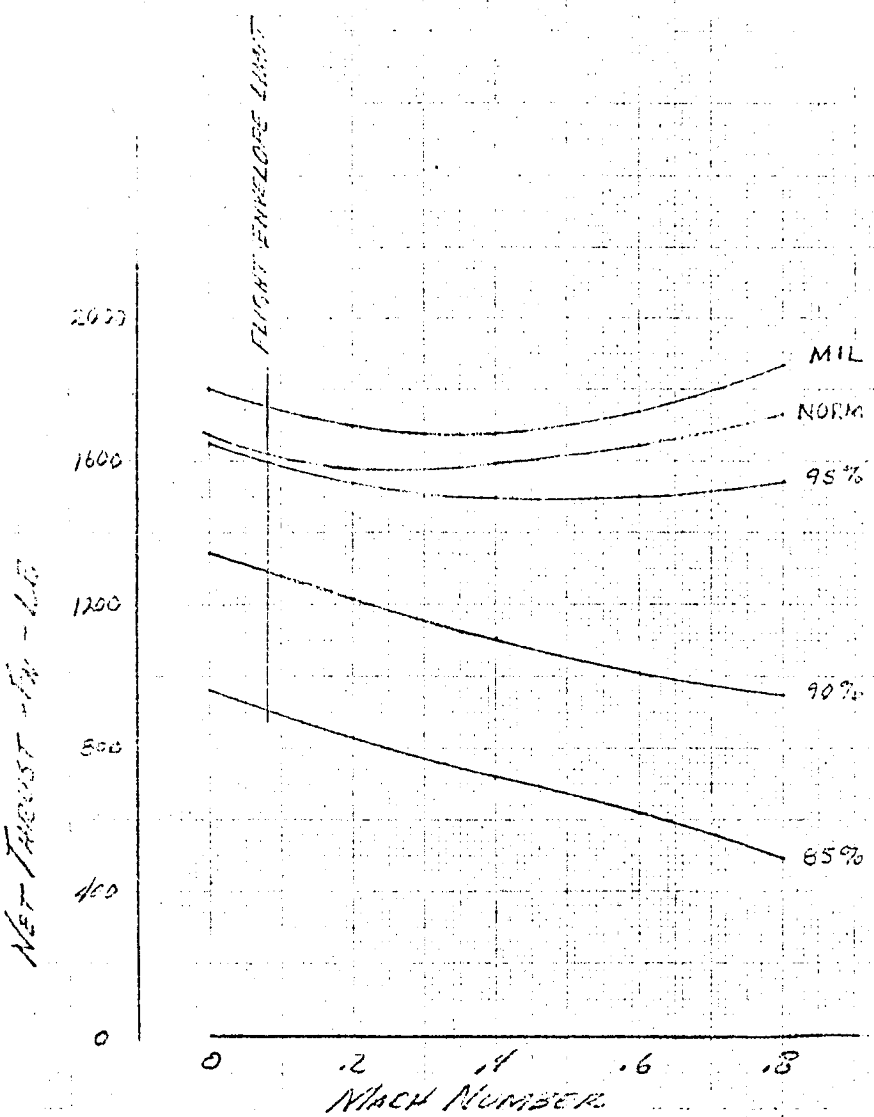


FIGURE 3-14. CRUISE THRUST VERSUS MACH NUMBER AND POWER SETTING FOR 15,000 FOOT STANDARD DAY DIVERTER VALVE +20 INCH TAILPIPE

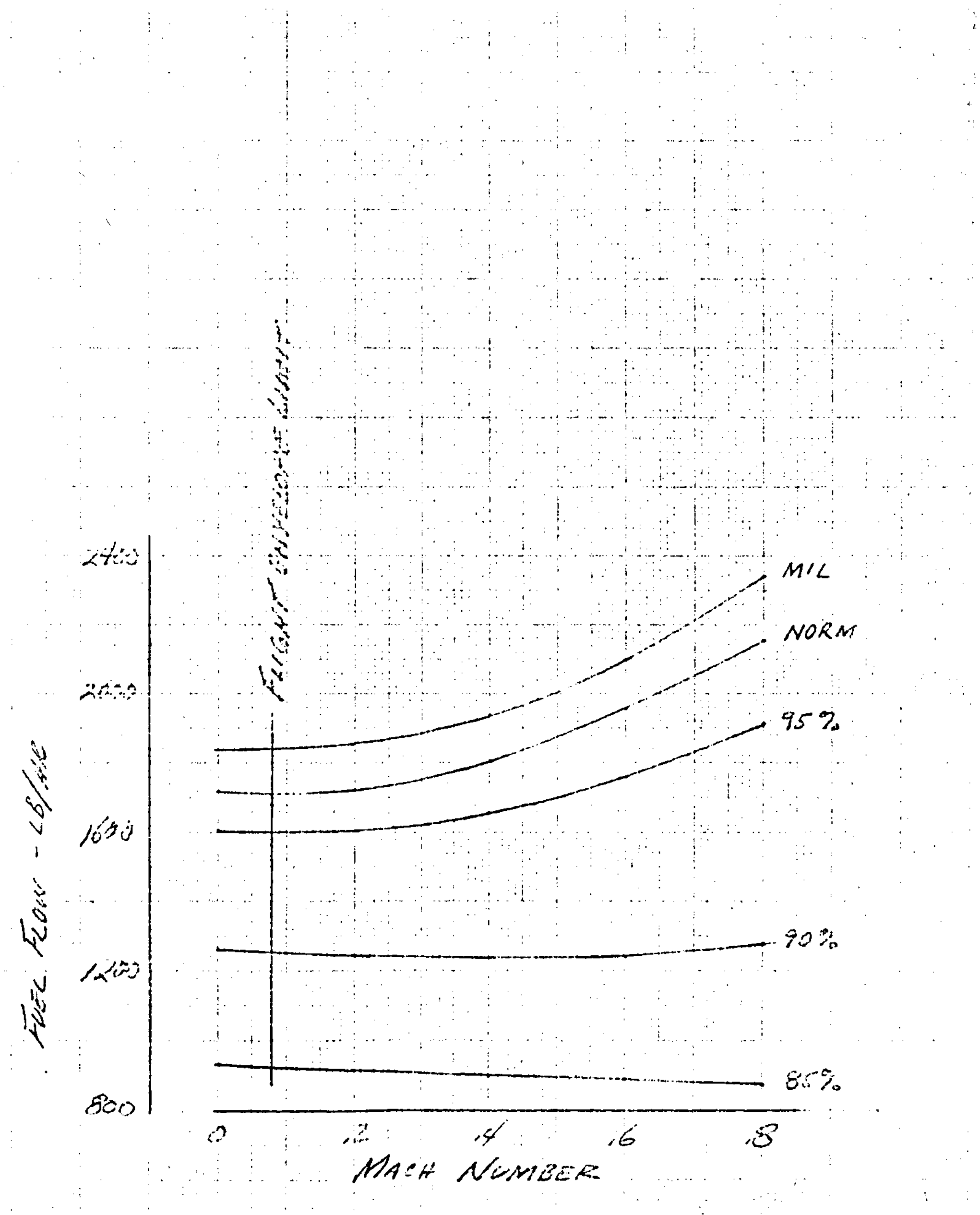


FIGURE 3-15. CRUISE FUEL FLOW VERSUS MACH NUMBER AND POWER SETTING - 15,000 FOOT STANDARD DAY DIVERTER VALVE +20 INCH TAILPIPE

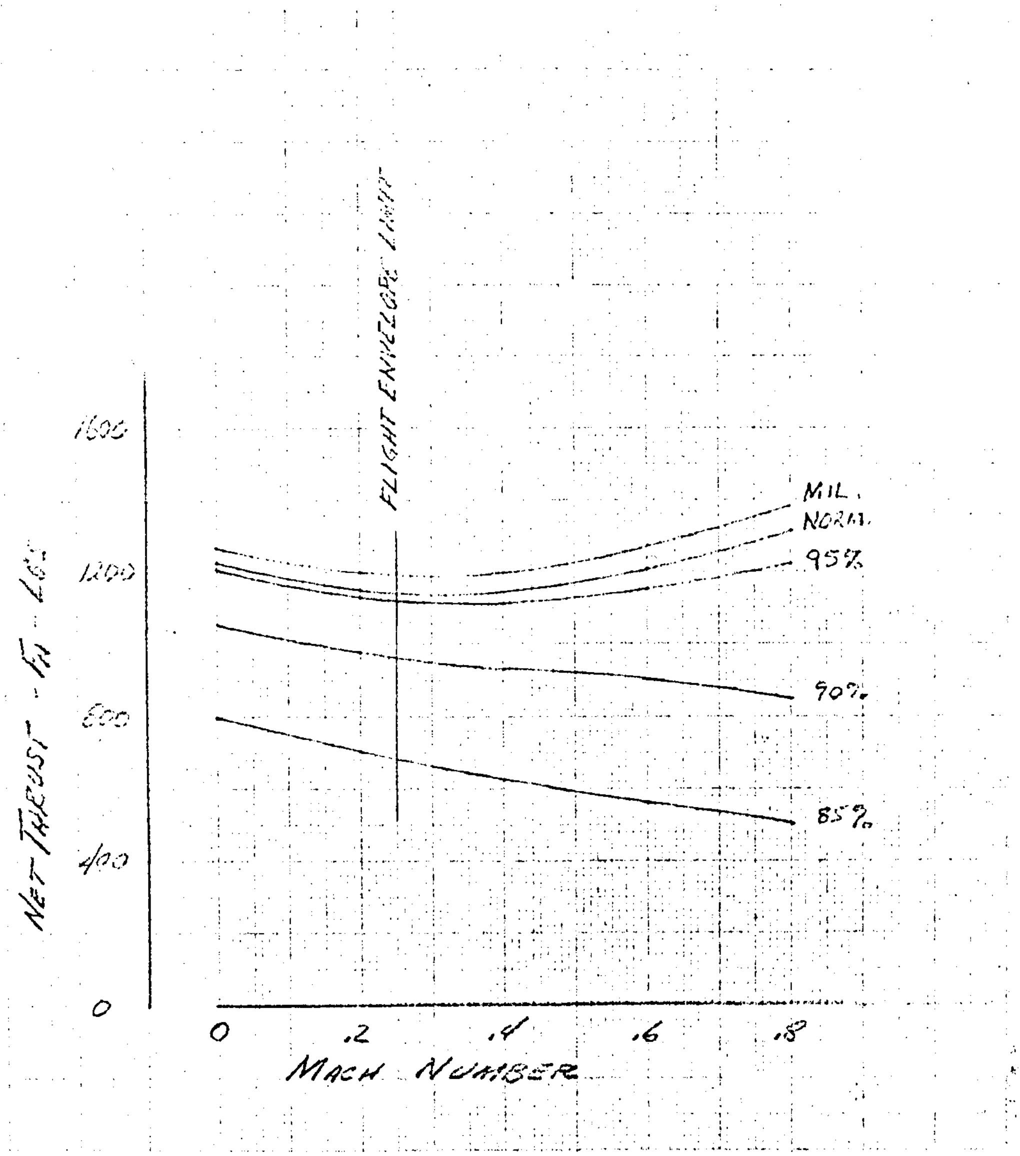


FIGURE 3-16. CRUISE THRUST VERSUS MACH NUMBER AND POWER SETTING FOR 25,000 FOOT STANDARD DAY DIVERTER VALVE +20 INCH TAILPIPE

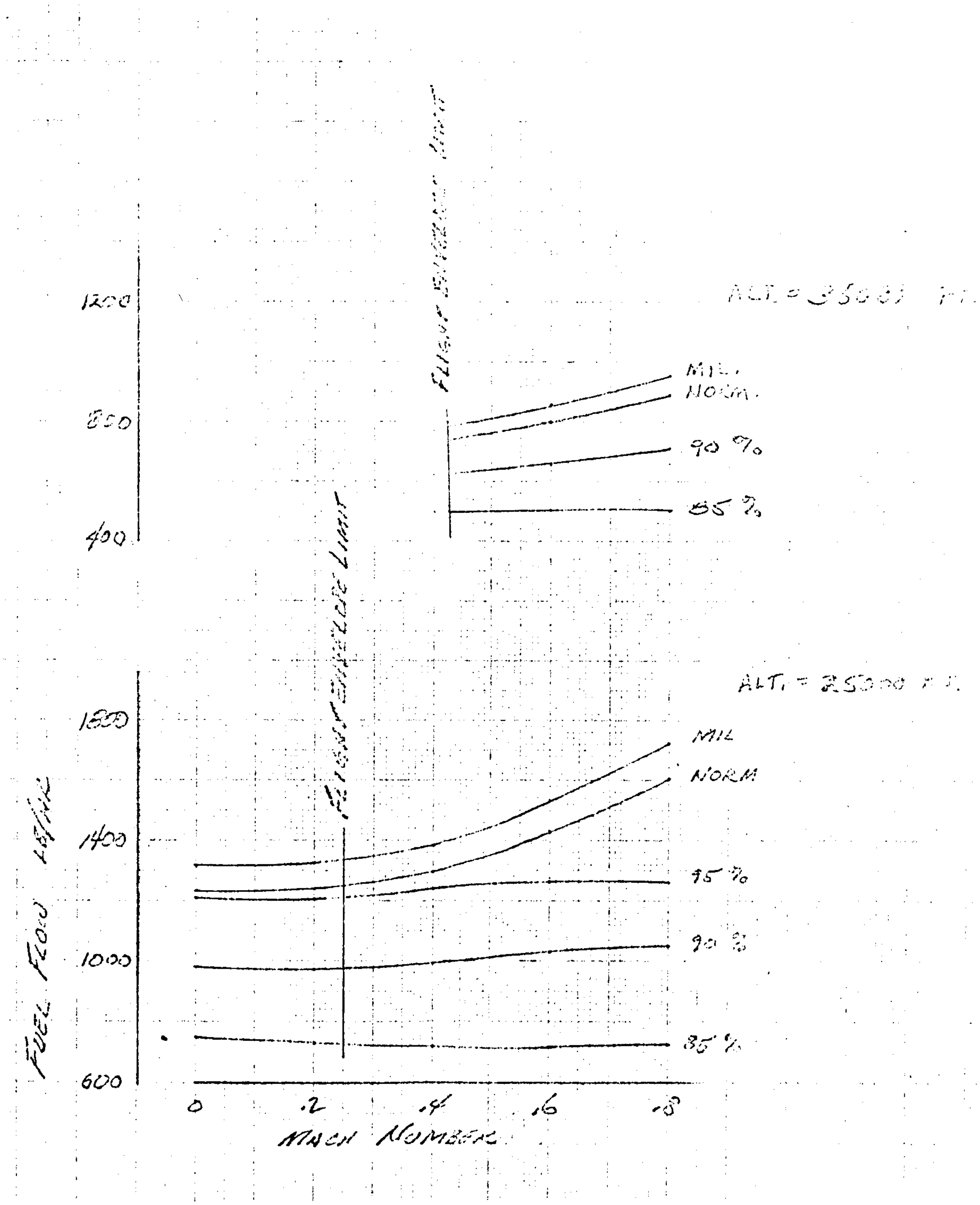


FIGURE 3-17. CRUISE FUEL FLOW VERSUS MACH NUMBER AND POWER SETTING - 25,000 AND 36,089 FOOT STANDARD DAY DIVERTER VALVE +20 INCH TAILPIPE

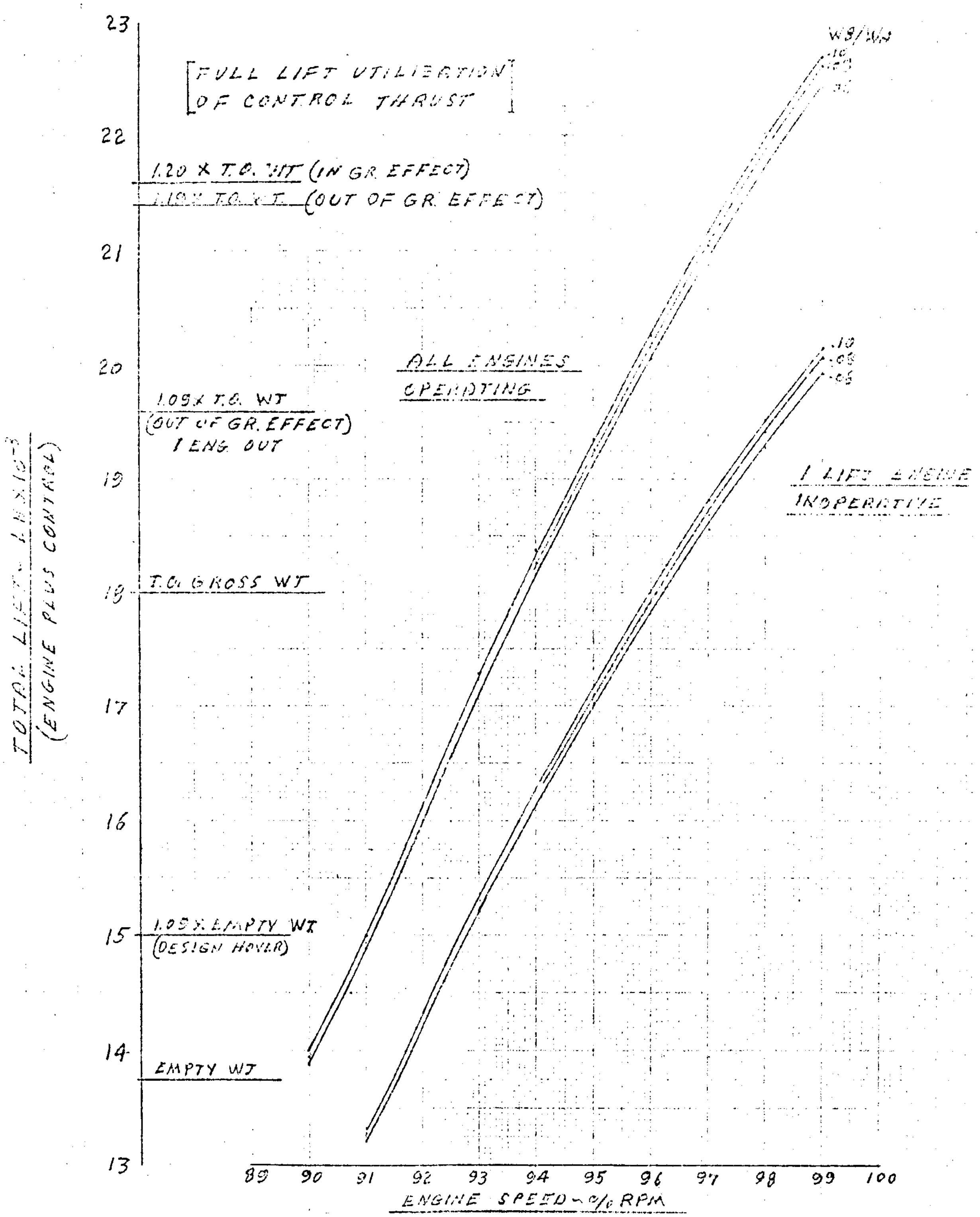


FIGURE 3-18. LIFT CAPABILITY NEW AIRCRAFT, N-309 COMPOSITE LIFT MODE 7 LIFT, 2 L/C YJ85-19 ENGINES SEA LEVEL 80 F

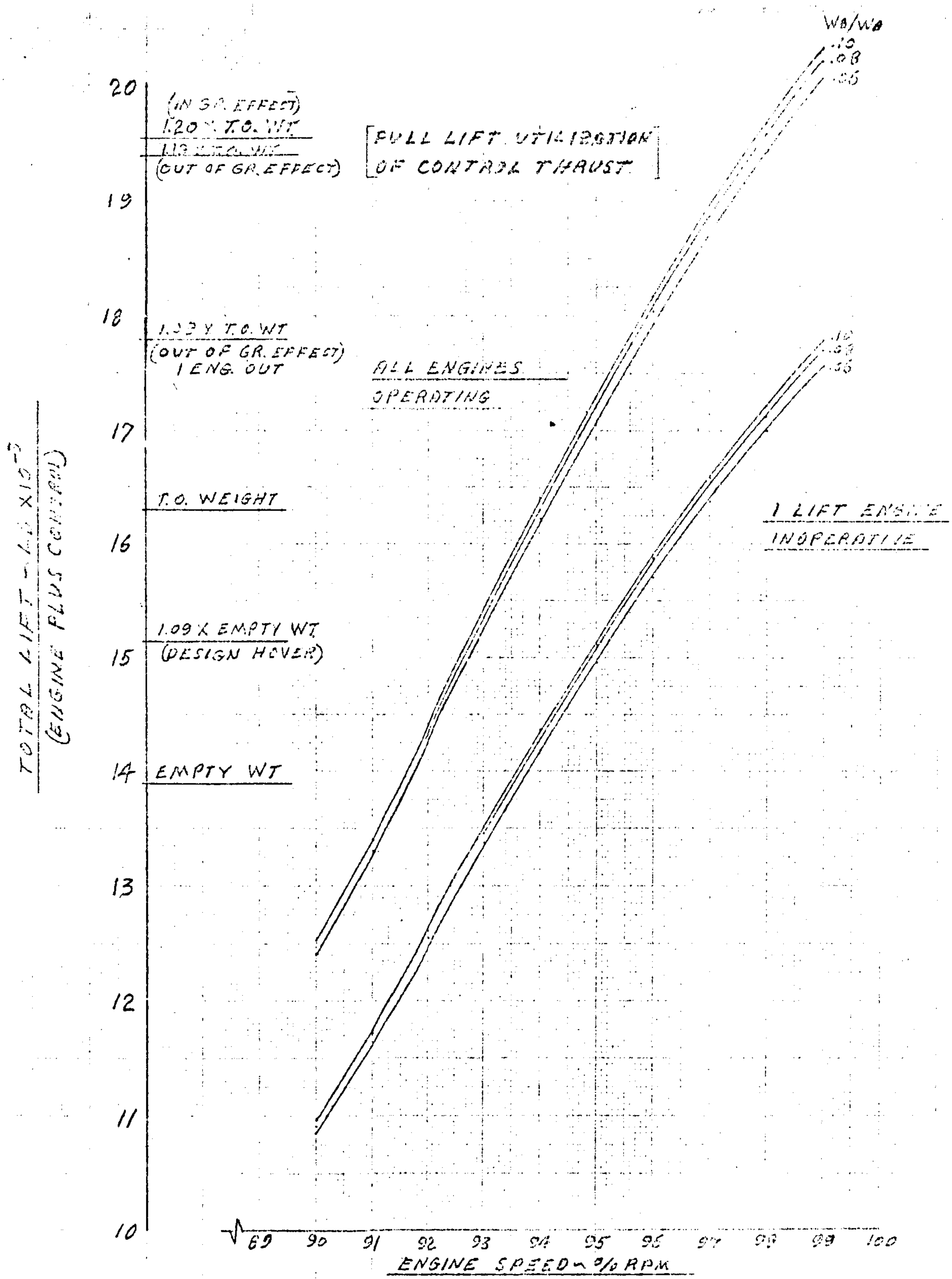


FIGURE 3-19. LIFT CAPABILITY NEW AIRCRAFT, N-309 DIRECT LIFT MODE 8 LIFT YJ85-19 ENGINES SEA LEVEL, 80 F

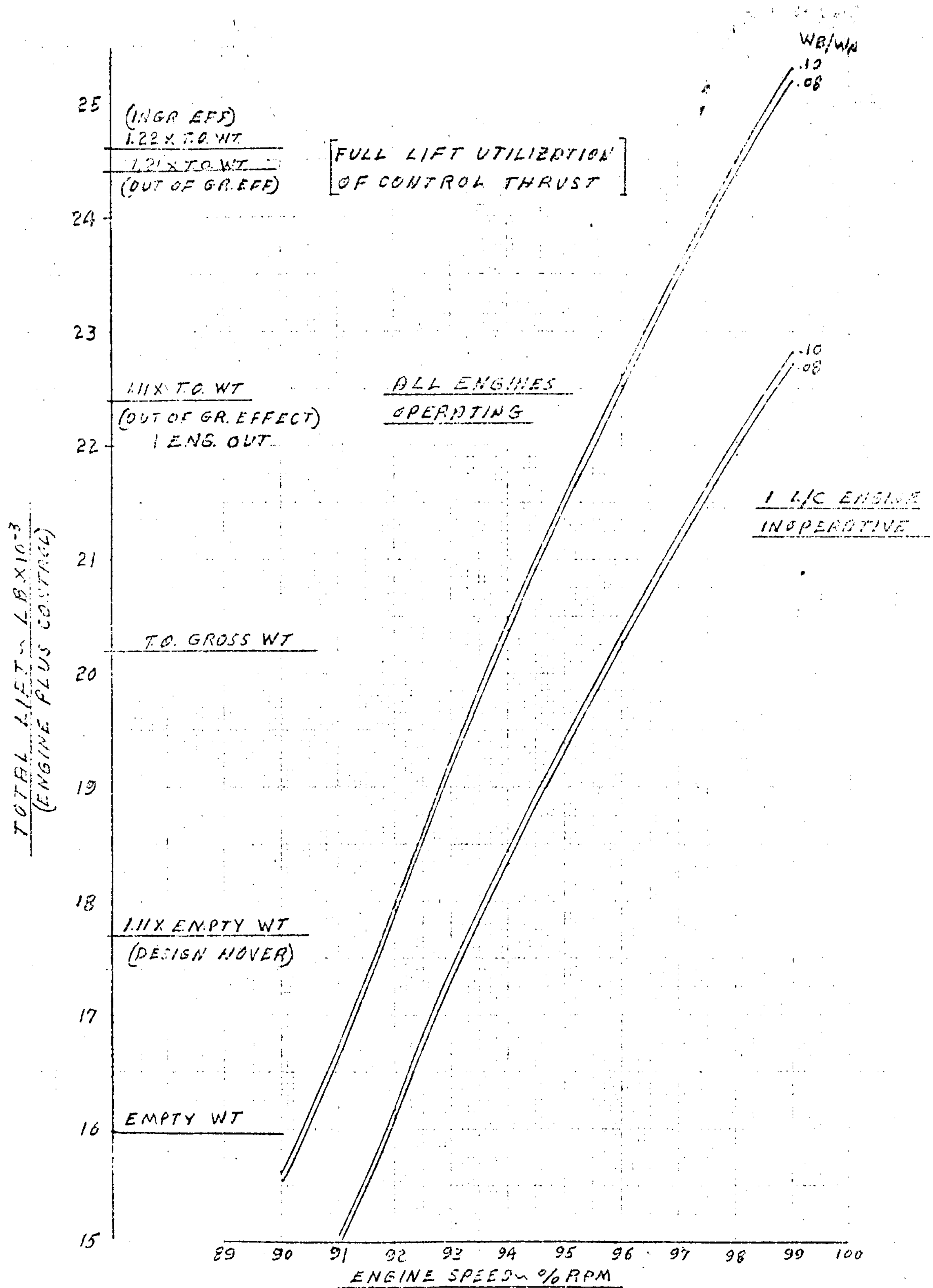


FIGURE 3-20. LIFT CAPABILITY MOD T-39A AIRCRAFT COMPOSITE LIFT MODE 8 LIFT, 2 L/C YJ85-19 ENGINES SEA LEVEL, 80 F

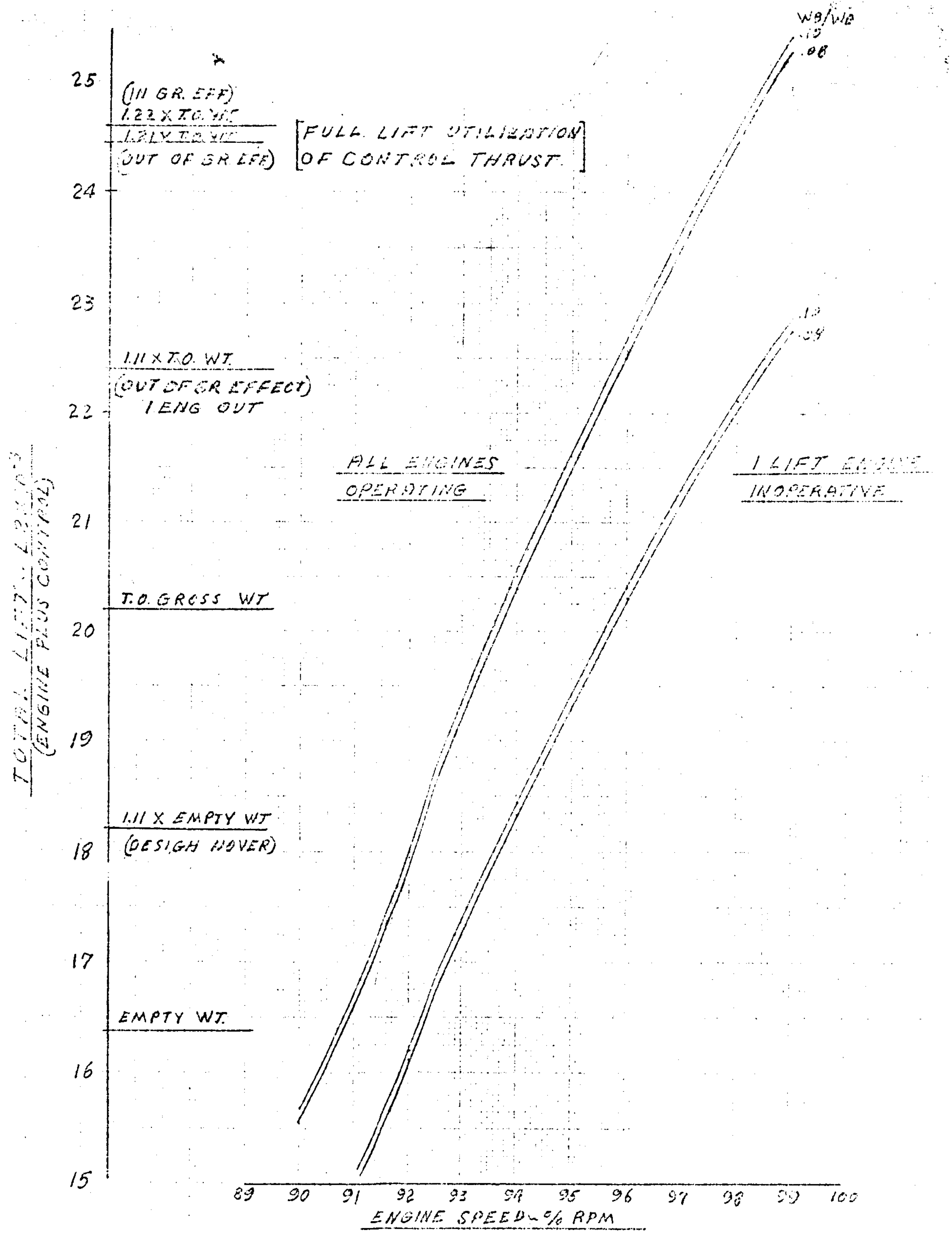


FIGURE 3-21. LIFT CAPABILITY MOD T-39A AIRCRAFT DIRECT LIFT MODE 10 LIFT J85-19 ENGINES SEA LEVEL, 80 F

operation at transitional flight speeds. The first objective is easily met with a bell-mouth shaped inlet with a radius of about one-half the diameter. The second objective, however, is not easily solved due to the extreme range of inlet velocity ratio imposed on the inlet. As shown by Figure 3-22, inlet velocity ratios at 200 knots vary from 7.0 at the minimum start speed of 12% rpm, down to 3.1 at idle speed (N = 47% rpm), and to below 1.0 at high power settings.

Tests of VTOL lift-engine plain inlets and scoop-type doors were conducted by Northrop Norair in the NASA/Ames 40-by-80 foot wind tunnel using the Ames lift engine pod containing five tandem-mounted YJ-85 engines. Figure 3-23 presents inlet recovery and distortion data from these tests as a function of inlet velocity ratio. No data with plain inlets at positive angles of attack were obtained. The performance shown on this figure at positive angles and "starting inlet velocity ratios are estimates only and might well be optimistic for some of the engines, particularly for the third bank and aft engine of the N-309 design.

The conclusion was that the performance of a plain bell-mouth inlet may not be satisfactory and that the design should incorporate devices to insure reliable performance throughout the entire operational range. Distortion at the compressor face at high inlet velocity ratios is the primary concern, not inlet recovery, since the air-impingement method is to be used to start the lift engines during flight. However, inlet pressure recoveries improve when distortion is decreased and any increased windmilling speed resulting from higher recovery decreases the time required for starting the lift engines.

The proposed inlet design consists of bell-mouth shaped inlets with internal flow turning vanes (Figure 3-24). The radius of the bell mouth for the forward pair of each group of lift engines varies from 10.0 inches (r/d = 0.625) at the most forward point to 6.0 inches (r/d = 0.375) at the aft centerline. For the second pair and single aft lift engine of the N-309 design, the forward radius is 8.0 inches (r/d = 0.5) tapering to 6.0 inches (r/d = 0.375) at the rear. The static recovery of this inlet is estimated at 0.995 at maximum engine power increasing to near unity at lower power settings.

Alternate designs, also shown in Figure 3-24, feature either external turning vanes or scoop-type doors. Both of these systems have been tested with some success in improving recovery and reducing distortion. However, they are more complex than the "fixed" internal vane system and, in the case of the scoop-type door, impose

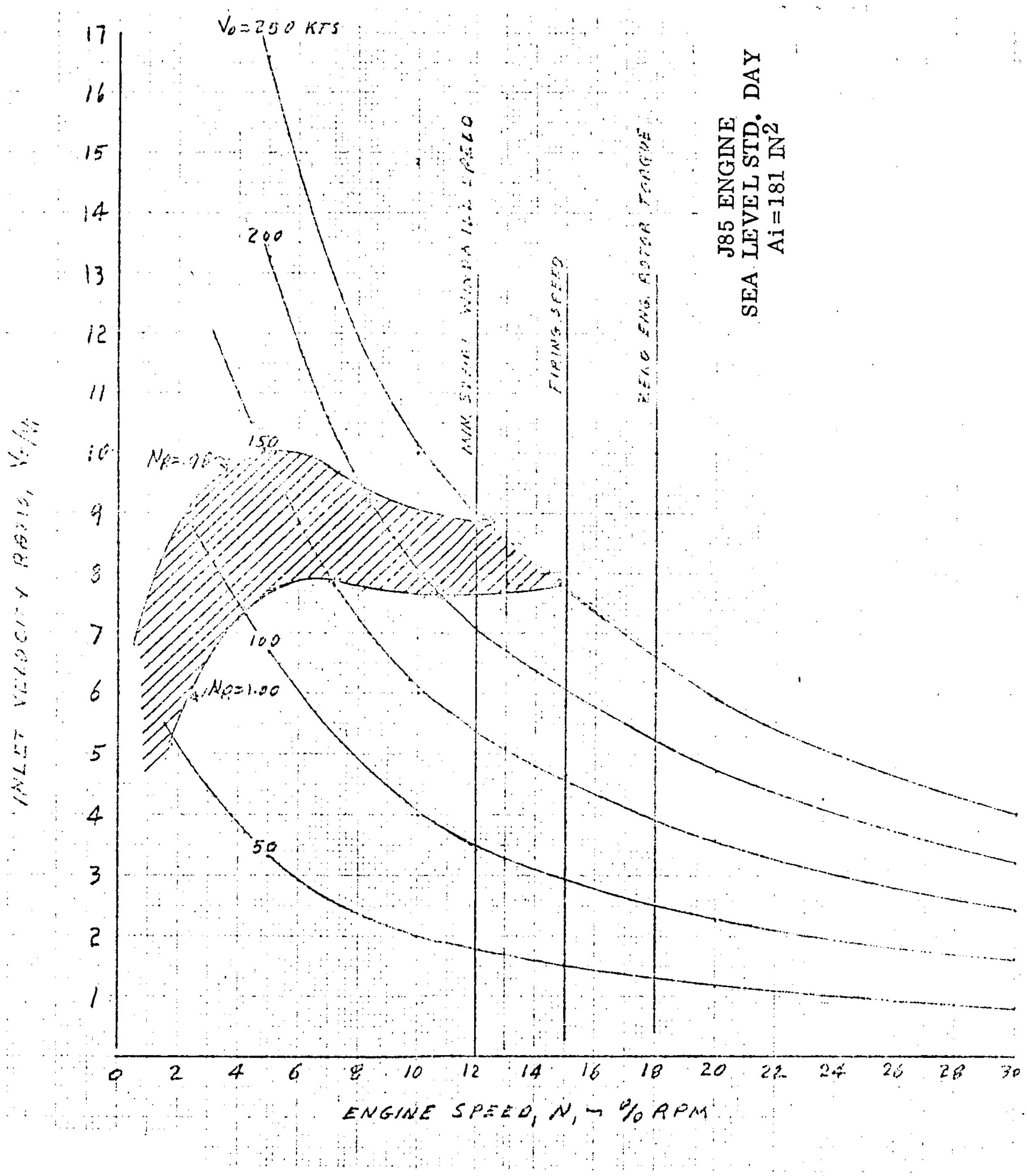


FIGURE 3-22. INLET VELOCITY RATIO VS ENGINE SPEED (LOW ENGINE SPEED)

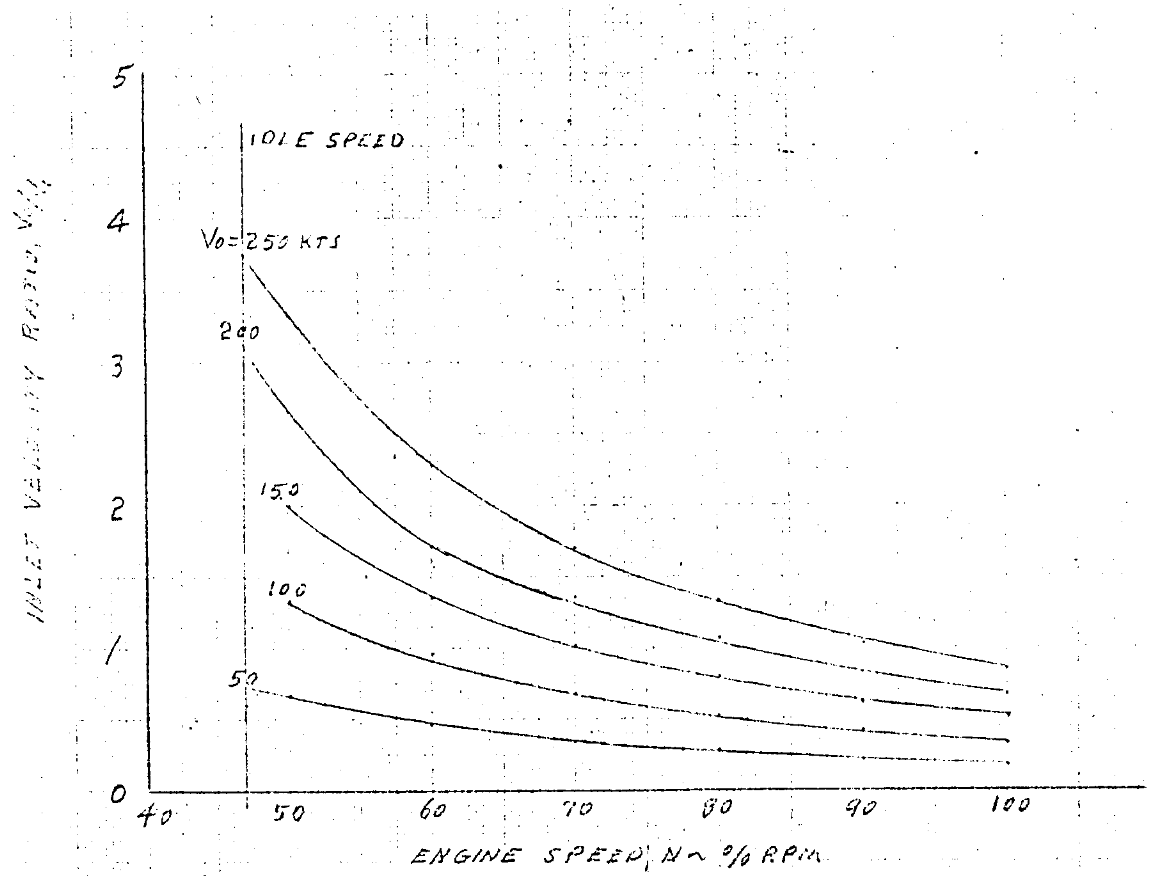


FIGURE 3-22A. INLET VELOCITY RATIO VS ENGINE SPEED (HIGH ENGINE SPEEDS)

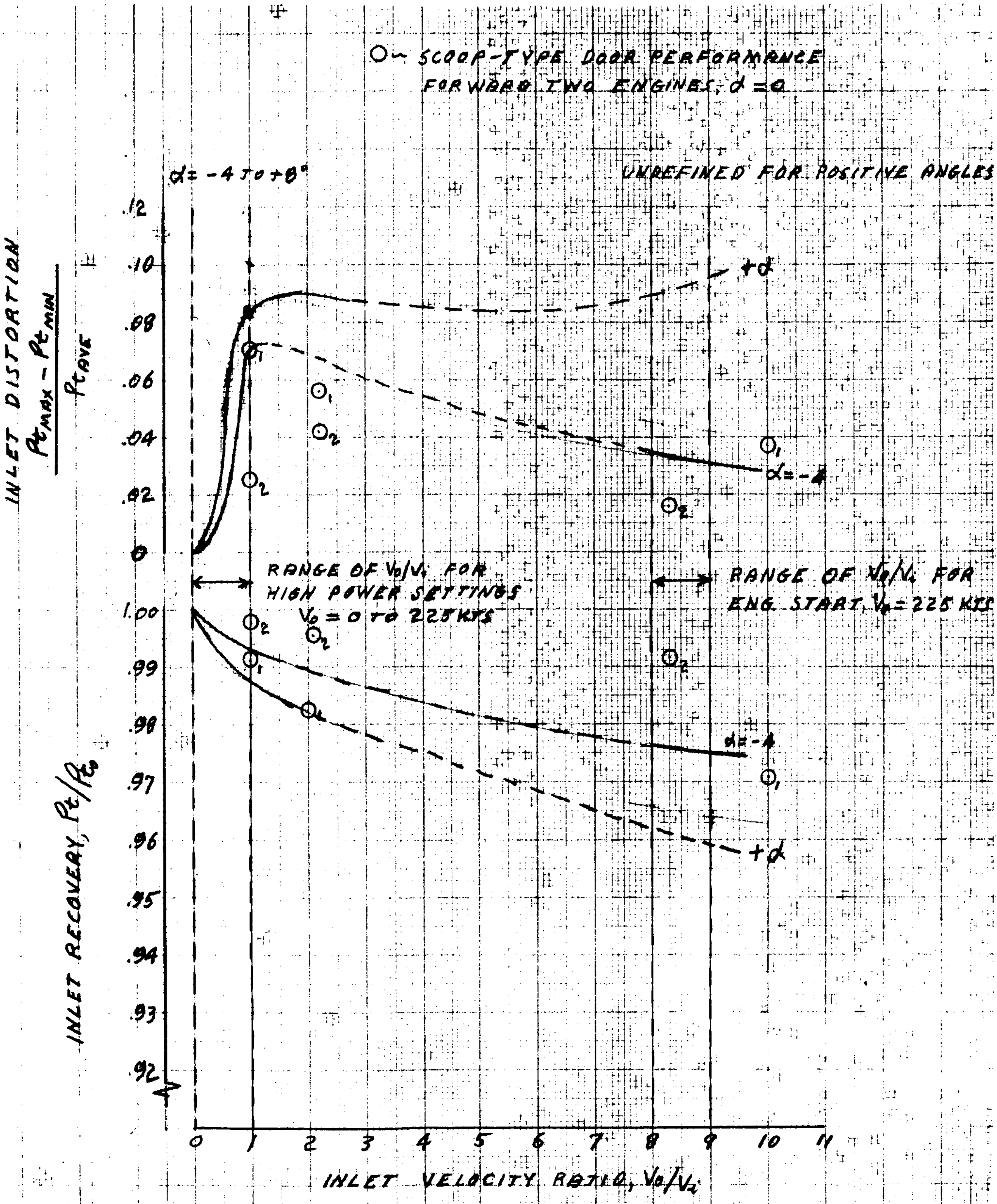
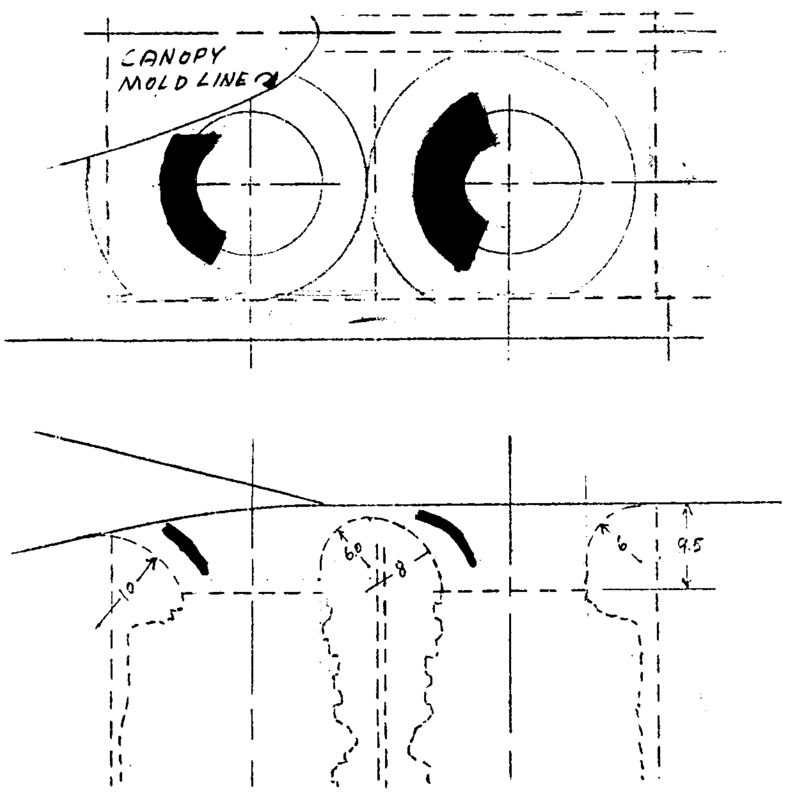


FIGURE 3-23. ESTIMATED YJ85-19 LIFT-ENGINE PLAIN INLET PERFORMANCE

DESIGN OBJECTIVE — MINIMIZE COMPRESSOR FACE DISTORTION FOR ENGINE START, IDLE OPERATION AND ACCELERATION AT TRANSITION FLIGHT SPEEDS

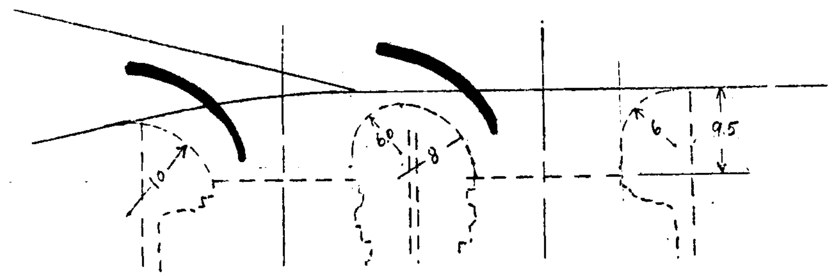
PROPOSED DESIGN

(Internal Turning Vanes)



ALTERNATE DESIGNS

External Turning Vanes



Scoop-Type Inlet Doors

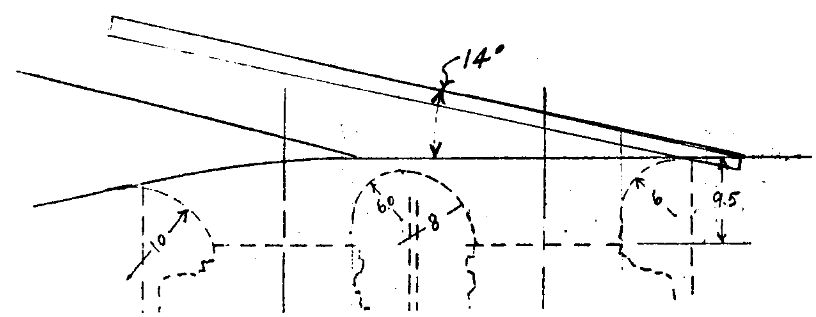


FIGURE 3-24. LIFT ENGINE INLET DESIGN

high drags during the engine start and idle phases at high transitional speed. Also, either a variable door system or suck-in doors would be required to obtain a favorable match between engine mass flow ratio and inlet area.

Tunnel testing of the proposed internal-vane system is required to substantiate performance and of the other systems if it should prove unsatisfactory.

3.7.2 Lift/Cruise Inlet

Design objectives were: (1) obtain a high static recovery at take-off power for maximum lift in the composite-lift mode; and (2) obtain high subsonic pressure recovery and a critical Mach number of about 0.80. The first objective dictated a short inlet with minimum diffusion to limit friction and diffuser losses, and a rounded or elliptical internal lip contour for favorable flow characteristics. The length of the inlet for the N-309 is 27.0 inches and no auxiliary inlet doors for static maximum power operation are required. The estimated recovery performance for static and flight operation is shown in Figure 3-25.

Actual inlet profile coordinates have not been established. However, profiles specified for the NACA 1-45-165 elliptical inlet appear to offer the best compromise between static and flight operation in the subsonic range.

3.8 THRUST VECTORING

The new design (N-309) and the Mod. T-39A incorporate single-plane spherical vectoring nozzles on all lift engines and a combination vectoring-closure door below the lift exhaust nozzle of each lift/cruise engine.

The single-plane vectoring nozzle will be manufactured by General Electric and is a fall-out of their current development of a two-plane spherical vectoring nozzle.

The single-plane vectoring nozzle has a fore-and-aft vectoring capability of 28 degrees and, as shown in Figure 3-26, is highly efficient in turning the exhaust. At full vectoring the thrust loss is only 4% of the unvectored thrust. Unvectored, the thrust loss is 1.5% relative to the basic YJ85-19 convergent nozzle.

The vectoring door (Figure 3-26) is not as efficient as the nozzle with a 12% thrust loss at an effective vectoring angle of 28 degrees. The 10-degree rearward cant of the lift/cruise engine exhaust nozzles results in a horizontal thrust component of 390 pounds per engine. However, an airplane pitch attitude change of only

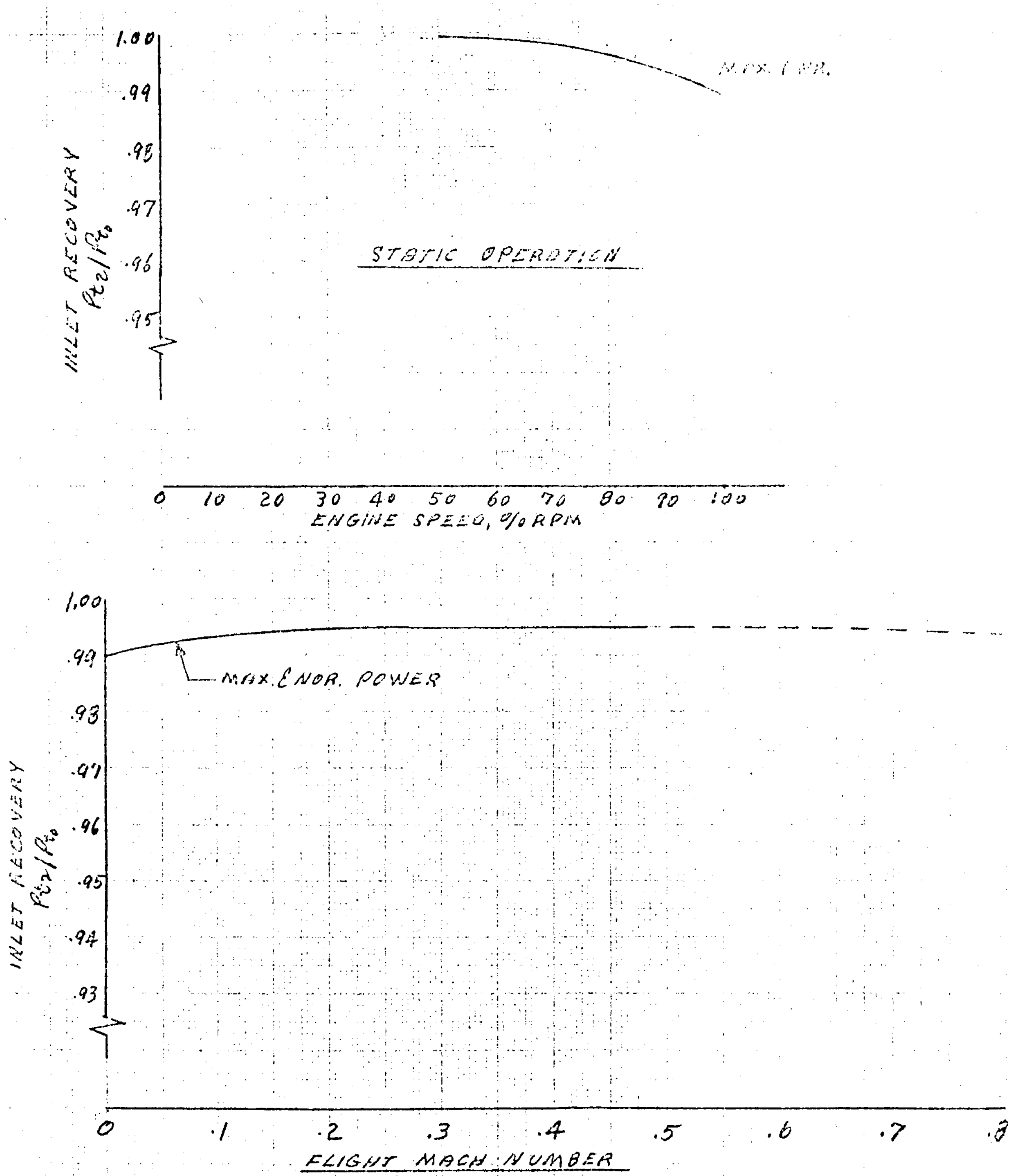


FIGURE 3-25. ESTIMATED LIFT/CRUISE ENGINE INLET PERFORMANCE

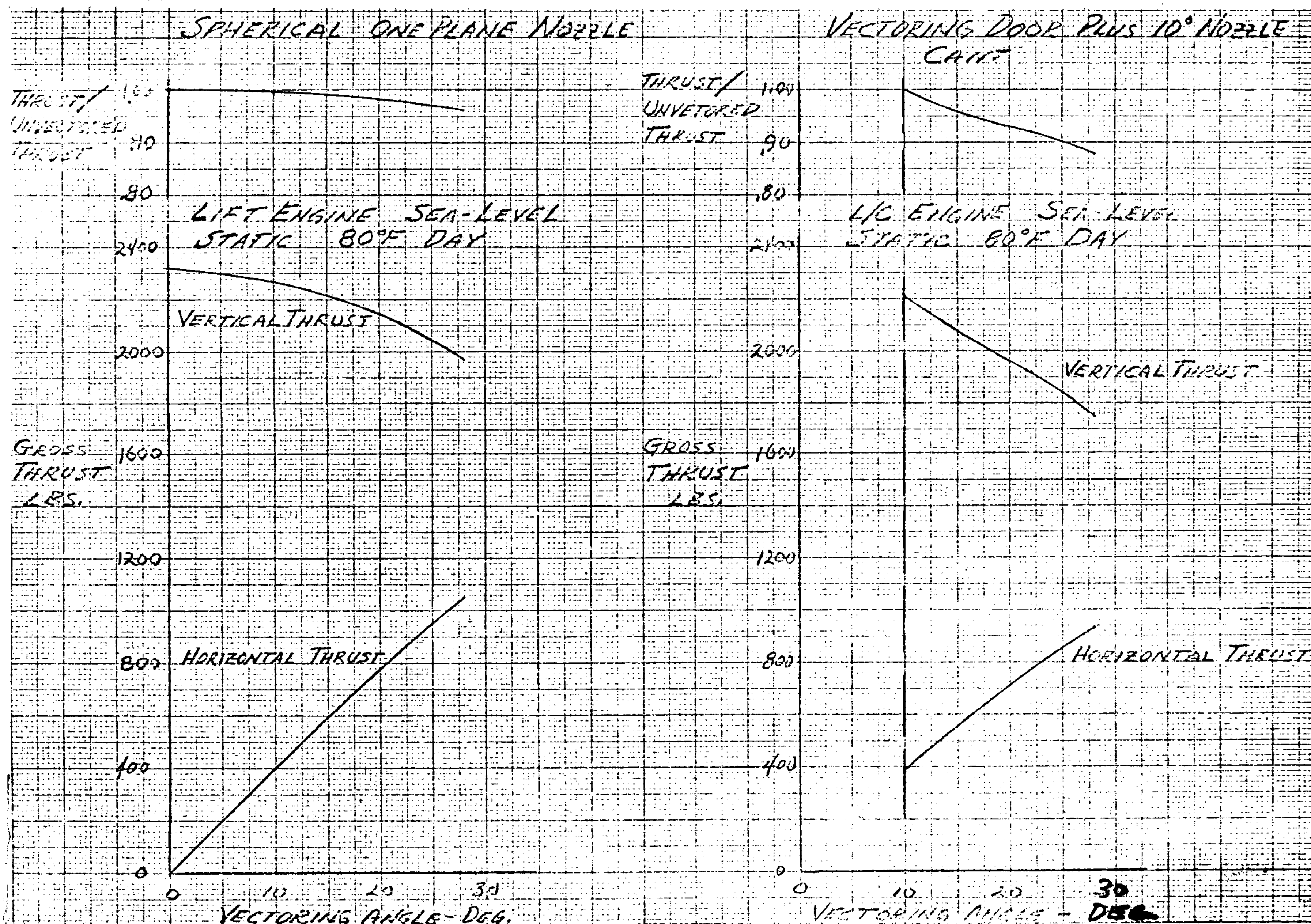


FIGURE 3-26. PROPULSION THRUST VECTORING EFFICIENCY NEW A/C

1.5 to 2.0 degrees is required to counteract this forward thrust with negligible effect on lift capability.

Overall, the vectoring door system results in about a 14-pound weight reduction due mainly to the elimination of the additional separate closure doors, which are required when vectoring nozzles are used. The ±28° vectoring nozzle is a slight deviation from the revised specification which specified vectoring of 15° forward and 30° aft.

In case of a requirement for more vectoring capability, an alternate vectoring configuration for the lift/cruise engines utilizing the same spherical nozzles employed on the lift engine was designed for the N-309 in the composite-lift mode (Figure 3-27, AD 4507). As indicated the design change is limited to the internal arrangement of the aft engine bay, aft fuel cell and reaction control duct routing. This alternate nozzle configuration still maintains a versatile capability to convert to the direct lift mode by removal of the lift/cruise exhaust system and the installation of another lift engine. The reaction control duct routing for pitch and yaw (on drawing AD-4486) is directed along the lower fuselage center line for the length of the airplane. However the spherical nozzle requires more longitudinal space for movement and activating mechanism. Therefore, to make efficient use of the space available the reaction control duct in this area is directed vertically, then aft, past the lift engine to the aft control nozzle. The shape of the aft fuel cell is revised to comply with the new duct routing. There is no external surface change to the configuration to accommodate the installation of the sperical nozzles.

3.9 LIFT ENGINE IN-FLIGHT STARTING

Lift engine starts during transition will be accomplished by boosting windmilling speed to start speed by turbine air impingement using bleed air from the lift/cruise engines. The bleed-air flow path is from the engine compressor into the main attitude control duct, then through ducting into the "start" turbine ports of the lift engines. The attitude control nozzles and the compressor bleed ports of the lift engines are closed.

Air-impingement requirements for static starts of a single engine using an external air supply (start cart) are as follows: (1) an airflow of 1.7 pounds/per second; (2) an air pressure of 43.0 psia; and (3) an air temperature of 820°R. As shown in Figure 3-28, a bleed air flow of 6.2 to 7.4 pounds/per second at a pressure level

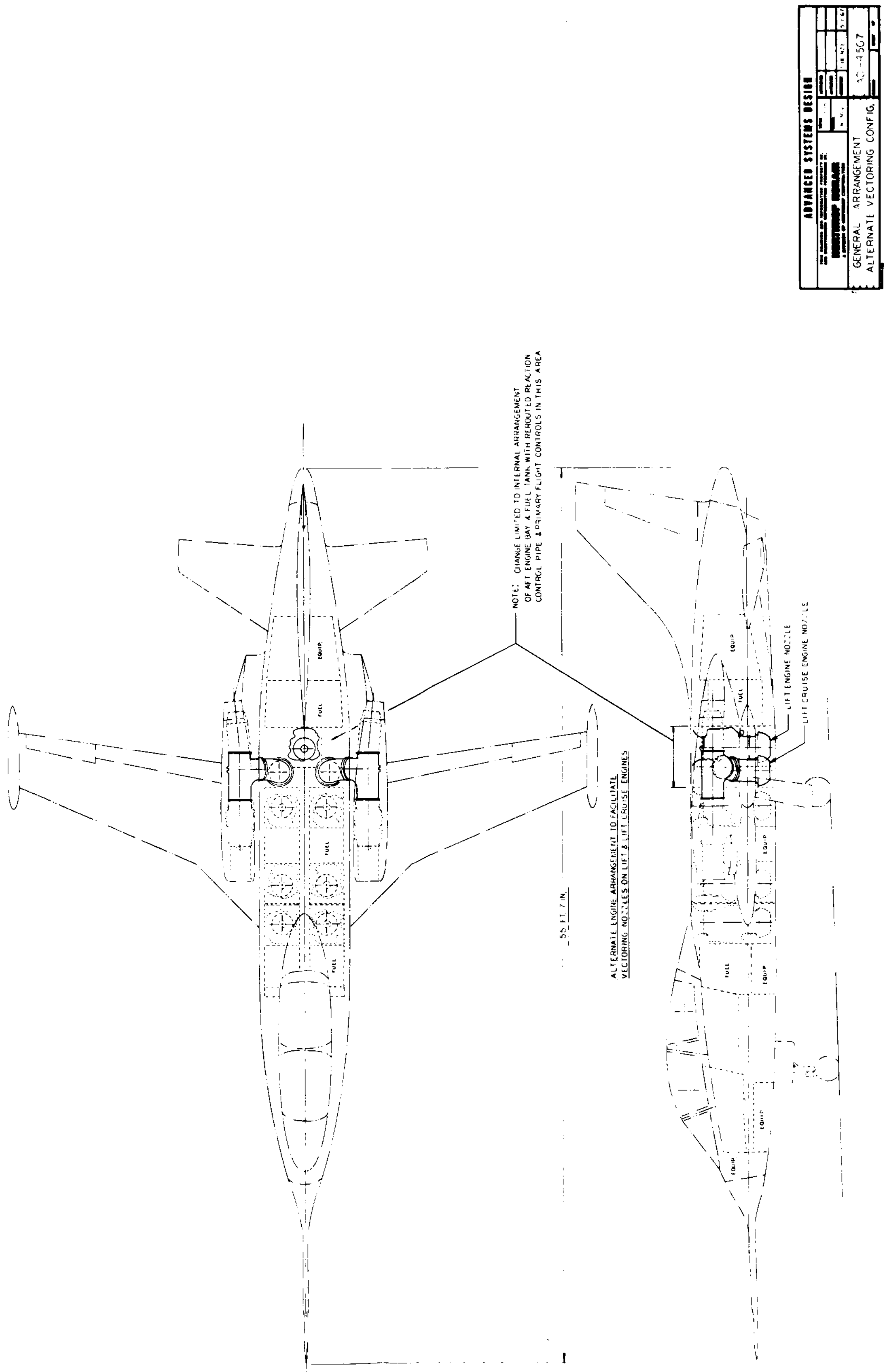

ADVANCED SYSTEMS DESIGN
GENERAL ARRANGEMENT
ALTERNATE VECTORING CONFIG.
NOTE: CHANGE LIMITED TO INTERNAL ARRANGEMENT OF AFT ENGINE BAY & FUEL TANK WITH REROUTED REACTION CONTROL PIPE & PRIMARY FLIGHT CONTROLS IN THIS AREA
ALTERNATE ENGINE ARRANGEMENT TO FACILITATE VECTORING NOZZLES ON LIFT & LIFT CRUISE ENGINES
55 FT. 7 IN.
LIFT ENGINE NOZZLE
LIFT CRUISE ENGINE NOZZLE
FUEL
EQUIP

FIGURE 3-27

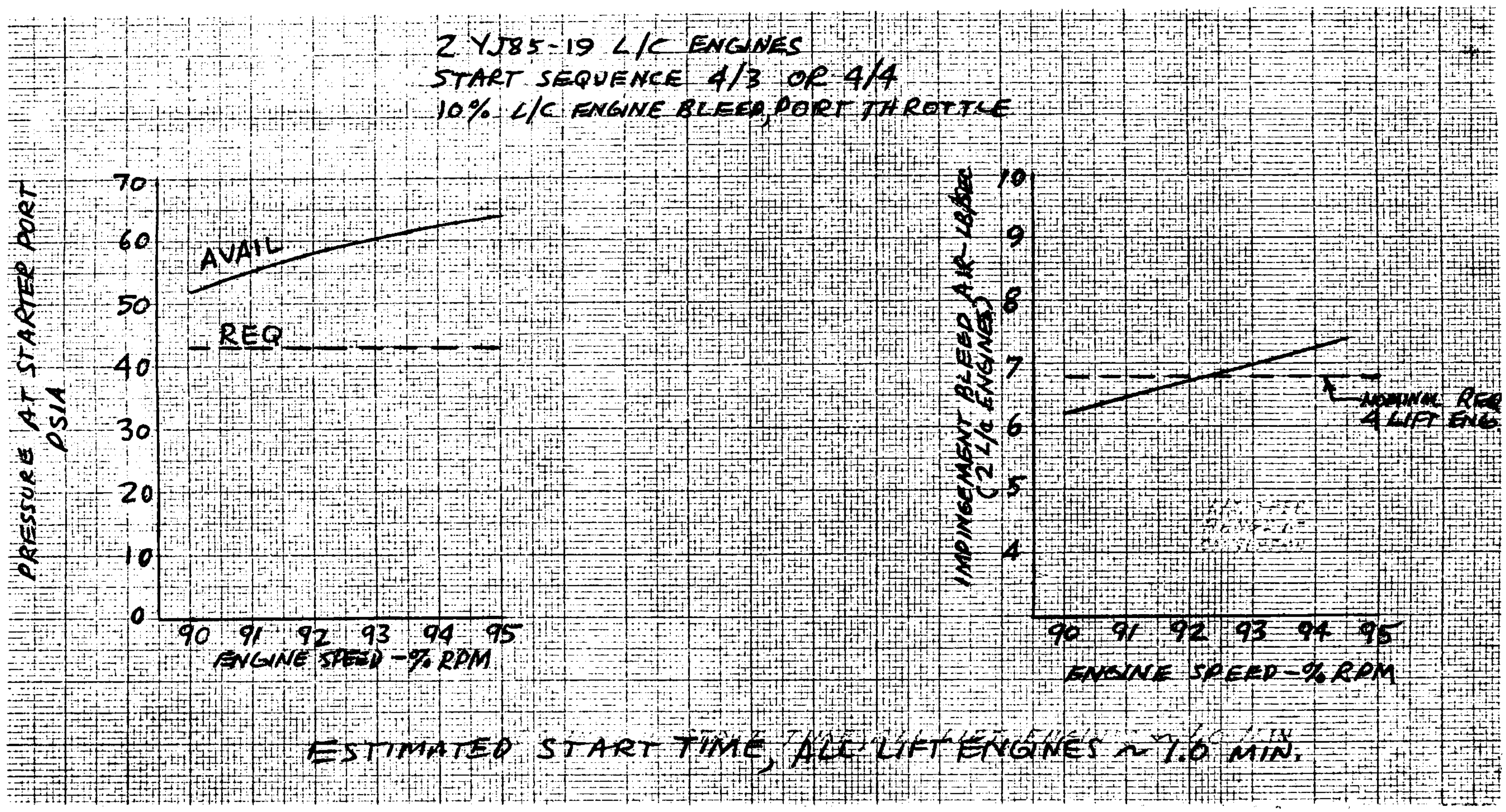


FIGURE 3-28. PROPULSION - LIFT ENGINE IMPINGEMENT START CAPABILITY

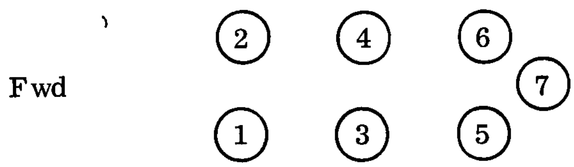
between 52 and 64 psia is available from two lift/cruise engines when bleeding the maximum 10%. A system pressure loss of 15.0% was used in the calculation of pressure at the starter port. The temperature of the bleed air varies from 885 R at 90% rpm to 945 at 94% rpm.

Specific horsepower (horsepower per pound of airflow) available from the lift/cruise engines is 100.5 compared with 72.6 from the "start" cart. Therefore, the available bleed-air energy from two lift/cruise engines is sufficient to start four lift engines simultaneously within 30 seconds. or 1.0 minute for the entire starting cycle.

In the cruise mode, the lift/cruise exhaust nozzle is sized for only a 1.0% bleed rate. Therefore, engine speed during the start sequence is restricted to a maximum of 94% rpm to limit the exhaust gas temperature to 1779 R, the maximum permissible for five minutes of operation. Engine thrust is reduced by about 5.5% when bleeding 10%, but is still adequate in the landing configuration at transitional flight speeds.

The lift-engine starting procedure for the N-309 airplane in the composite-lift mode is shown below, but is applicable to direct lift mode by the inclusion of a fourth engine in the second group of engines to be started. The starting procedure will be initiated at 200 knots. Engines are started in groups that minimize induced pitching and rolling moments with engines at idle speed. The engine starting sequence diagram is shown in Figure 3-29 (AD 4514).

N-309 Lift-Engine Starting Procedure



1. Open inlet and exhaust doors - single switch.

2. Move throttle to idle power setting.

3. Actuate single start switch:

   a. Air-impingement valves open and ignition energized on Engines 1, 2, 5 and 6.

   b. Fuel valves open at 12% rpm.

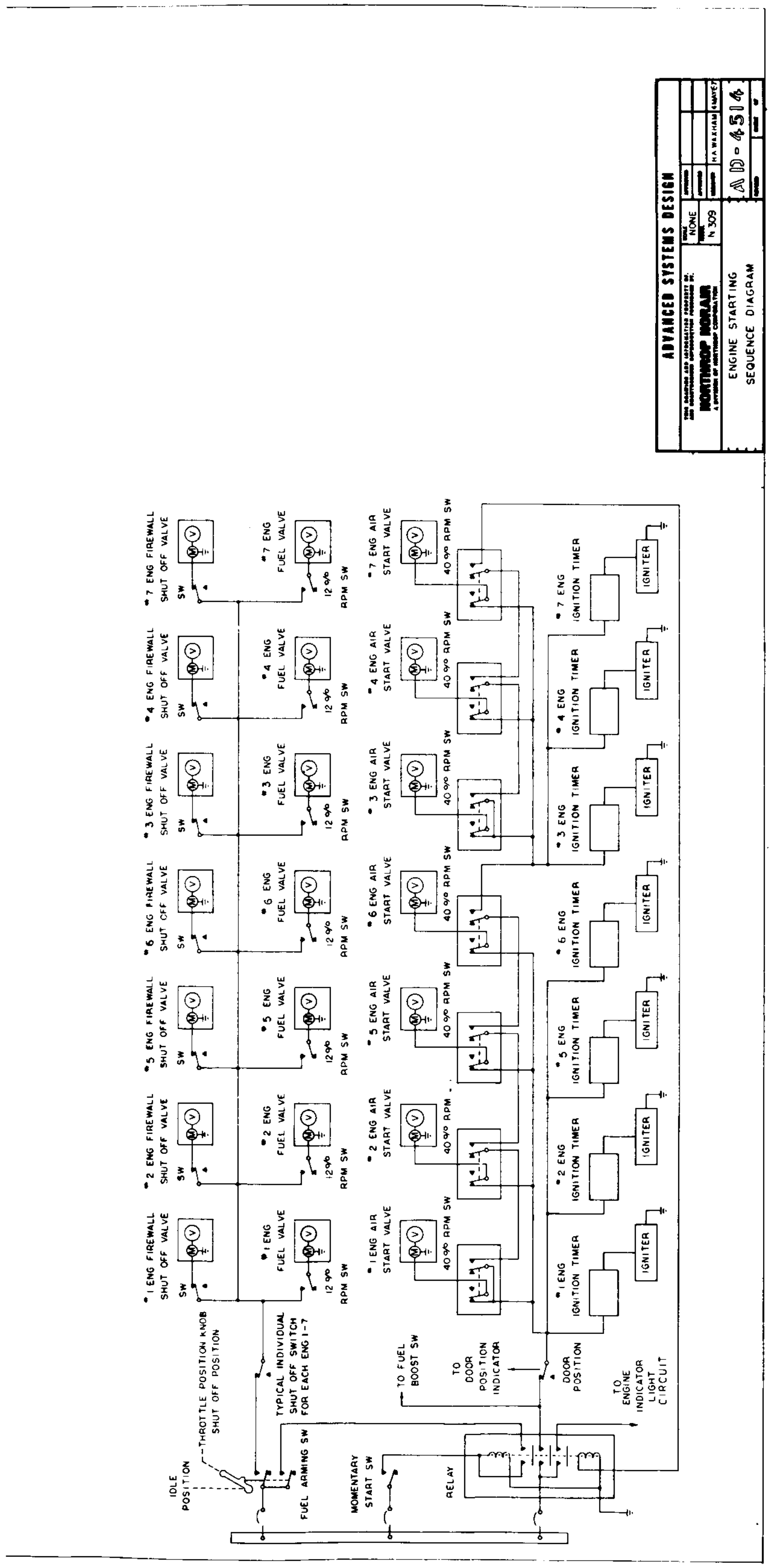

ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
ENGINE STARTING
SEQUENCE DIAGRAM
NONE
N 309
AD-4514
#1 ENG FIREWALL SHUT OFF VALVE SW
#2 ENG FIREWALL SHUT OFF VALVE SW
#5 ENG FIREWALL SHUT OFF VALVE SW
#6 ENG FIREWALL SHUT OFF VALVE SW
#3 ENG FIREWALL SHUT OFF VALVE SW
#4 ENG FIREWALL SHUT OFF VALVE SW
#7 ENG FIREWALL SHUT OFF VALVE SW
#1 ENG FUEL VALVE
#2 ENG FUEL VALVE
#5 ENG FUEL VALVE
#6 ENG FUEL VALVE
#3 ENG FUEL VALVE
#4 ENG FUEL VALVE
#7 ENG FUEL VALVE
12% RPM SW
#1 ENG AIR START VALVE
#2 ENG AIR START VALVE
#5 ENG AIR START VALVE
#6 ENG AIR START VALVE
#3 ENG AIR START VALVE
#4 ENG AIR START VALVE
#7 ENG AIR START VALVE
40% RPM SW
#1 ENG IGNITION TIMER
#2 ENG IGNITION TIMER
#5 ENG IGNITION TIMER
#6 ENG IGNITION TIMER
#3 ENG IGNITION TIMER
#4 ENG IGNITION TIMER
#7 ENG IGNITION TIMER
IGNITER
IDLE POSITION
THROTTLE POSITION KNOB SHUT OFF POSITION
TYPICAL INDIVIDUAL SHUT OFF SWITCH FOR EACH ENG 1-7
FUEL ARMING SW
MOMENTARY START SW
RELAY
TO FUEL BOOST SW
TO DOOR POSITION INDICATOR
DOOR POSITION
TO ENGINE INDICATOR LIGHT CIRCUIT

FIGURE 3-29

c. Air-impingement valves close at 40% rpm. Idle light OFF at 40% rpm and air-impingement valve closed, each engine.

d. Starting system for remaining engines armed when Engines 1, 2, 5 and 6 at idle speed and air-impingement valves closed.

4. Sequence (3) repeated for Engines 3, 4 and 7.

3.10 ENGINE PERFORMANCE SENSITIVITY

The effects on YJ85-19 engine performance of 50% variations in the installation losses are shown in Figure 3-30. The maximum additional thrust loss at take-off power (variations in the adverse direction) are 1.1% for the lift engine and 1.9% for the lift/cruise engine. The corresponding over-all loss of lift capability for the N-309, composite-lift mode, is 1.27%. The effect on the maximum available control thrust is to reduce it by 1.8%. The over-all specific fuel consumption is increased by 1.5% if all additional installations are 50% greater than estimated.

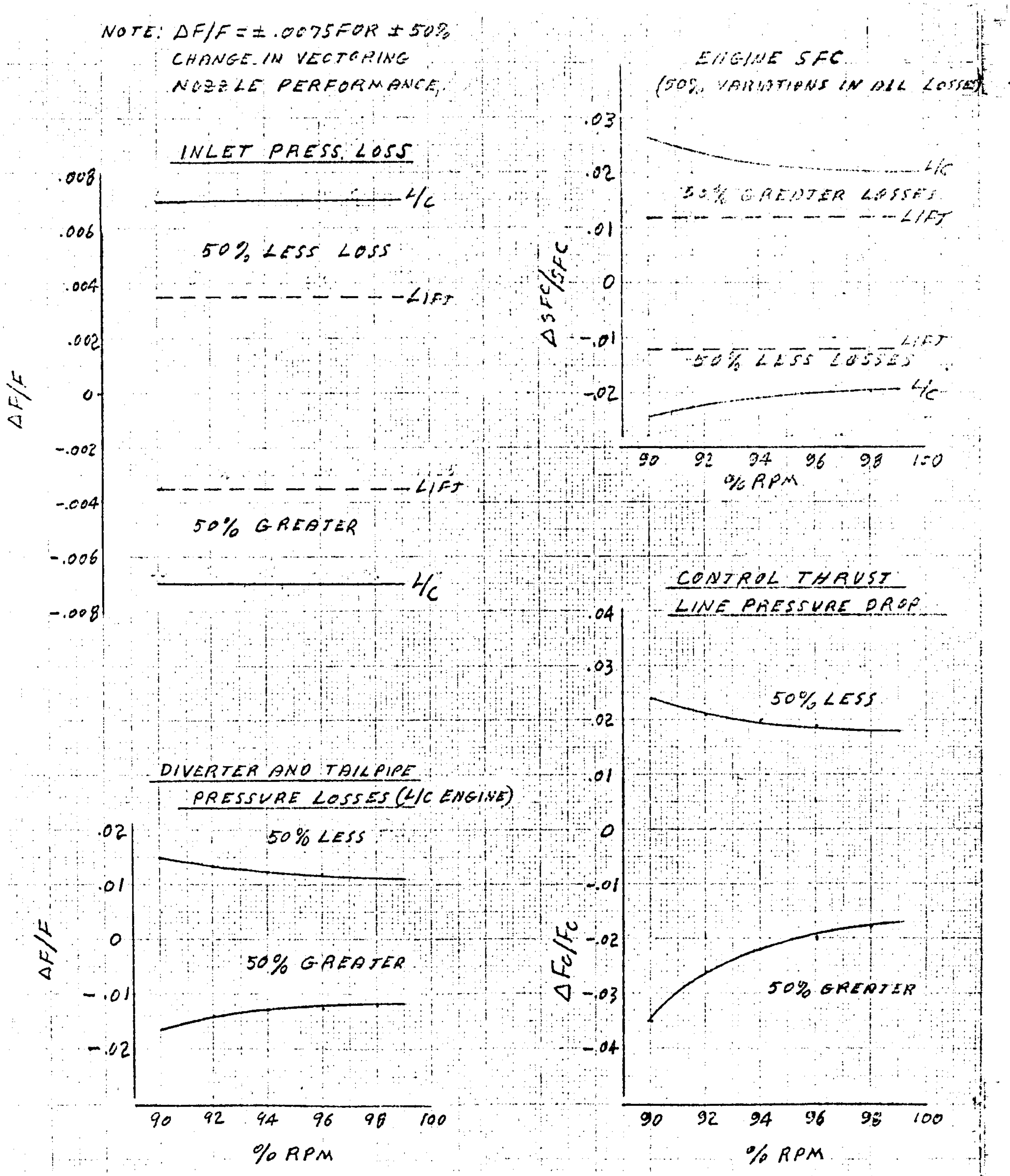


FIGURE 3-30. YJ85-19 ENGINE PERFORMANCE SENSITIVITY TO 50% VARIATIONS IN INSTALLATION LOSSES, STATIC OPERATION

4.0 REACTION CONTROL SYSTEM

4.1 GENERAL

In addition to providing the required control moments, design objectives of the reaction control system using bleed air from the engines were: (1) to obtain a mechanically simple system that eliminates the requirement for interaction between control nozzles when maximum control is demanded about a single axis; (2) minimize cross coupling between control axes and lift; and (3) minimize system weight.

A modified constant-bleed system was selected over a pure demand system to limit engine over temperature during extended test periods with probable high control demands. Other advantages were a faster response system and minimization of lift cross coupling when full control is demanded. Compressor bleed rates for the N-309 design in the composite lift mode and the Modified T-39A design varied from 8 percent for zero control demand to 10 percent, the engine limit. For the direct-lift mode of the N-309, a bleed-rate range of 6 to 10 percent was required to meet the control demands.

4.2 SYSTEM DESCRIPTION

Pitch and roll system ducting are interconnected to insure nearly equal control availability about either axis in case of failure of any engine. The flow areas of the ducts were selected to yield an initial flow Mach number of 0.30 to limit the pressure loss to 20.0 percent of the pressure available at the engine exit ports. Detailed pressure-drop analyses on the N-309 system substantiated the selected duct diameters at 8.5 in. for the pitch duct and 4.0 in. for roll with pressure drops of 20.0 percent occurring only when maximum control about a given axis is applied. To accommodate the increased flow from ten engines, duct diameters for the Mod. T-39A design are 9.6 in. for pitch and 4.5 in. for roll.

For pitch control, variable-area reaction nozzles are located at the extremities of the longitudinal axis. The nozzles exhaust downward only with differential nozzle reaction thrusts generating the required control moment.

Yaw control moment is obtained by rotating the pitch nozzles differentially until the required moment or attitude is established.

Reaction nozzles located at the wing tips and capable of exhausting either up or down provide roll control. This method reduces the size of the duct for inclusion within the wing mold line. For the N-309, composite-lift mode, both nozzles exhaust downward with differential nozzle thrusts providing roll control until one nozzle is completely open and the other closed. Additional control moment is obtained by progressively opening the upper nozzle of the previously closed nozzle. This procedure eliminates cross coupling between the applied roll moment and pitch and lift up to the application of about one-half of the available roll moment. For the N-309 direct-lift mode and the Mod T-39A design, a pure up-down system is employed; that is, one roll nozzle always exhausts in a direction opposite to the other. Nozzle area schedules for each airplane design are discussed under the appropriate heading.

Figures 4-1 and 4-2 show a preliminary layout of the reaction nozzle configuration for the pitch/yaw and the roll nozzles. The nozzles use materials of different thermal coefficients of expansion so that minimum leakage is achieved without the danger of binding. The gap is sized for zero bleed airflow and, thus, the differential expansion during thermal shock tends to increase leakage rather than binding the system. Teflon seal strip further minimizes leakage and friction under any load distortion.

Drawing AD-4499 and Figure 4-3 show reaction control duct routing installations for composite and direct lift cases respectively. Thin wall corrosion resistant steel and titanium are used to minimize weight of the ducting. Insulation and radiant foil are installed where required for protection of structure, equipment or personnel.

4.3 AVAILABLE CONTROL

4.3.1 General

The quality of the bleed air and the resulting control thrust available for bleed rates of 8 and 10 percent with and without system losses are shown in Figures 4-4 and 4-5 for the lift and lift/cruise engine, respectively. System losses consisted of nozzle leakage (3.0 percent), line pressure drop ($\Delta P/P = .15$ at 99 percent RPM), and a nozzle velocity coefficient of 0.96. Bleed air flows used for purposes other than control are also grouped with the system losses. These consist of 0.5 percent of compressor bleed air for cooling the engine bays and, for the lift/cruise engines only, an additional 0.5 percent for cockpit conditioning, The estimated discharge coefficient of the

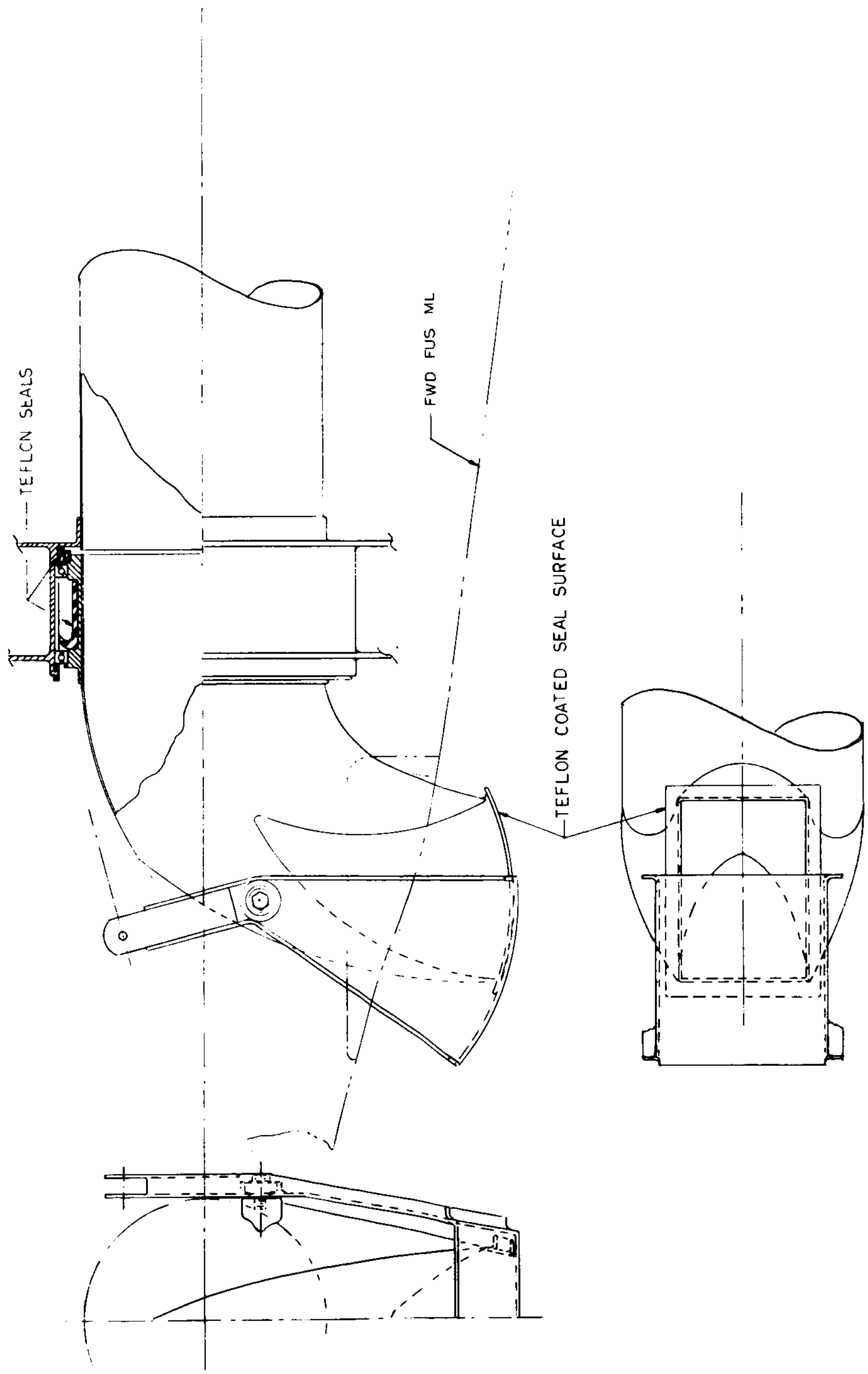


FIGURE 4-1. REACTION JET NOZZLE - YAW-PITCH CONTROL

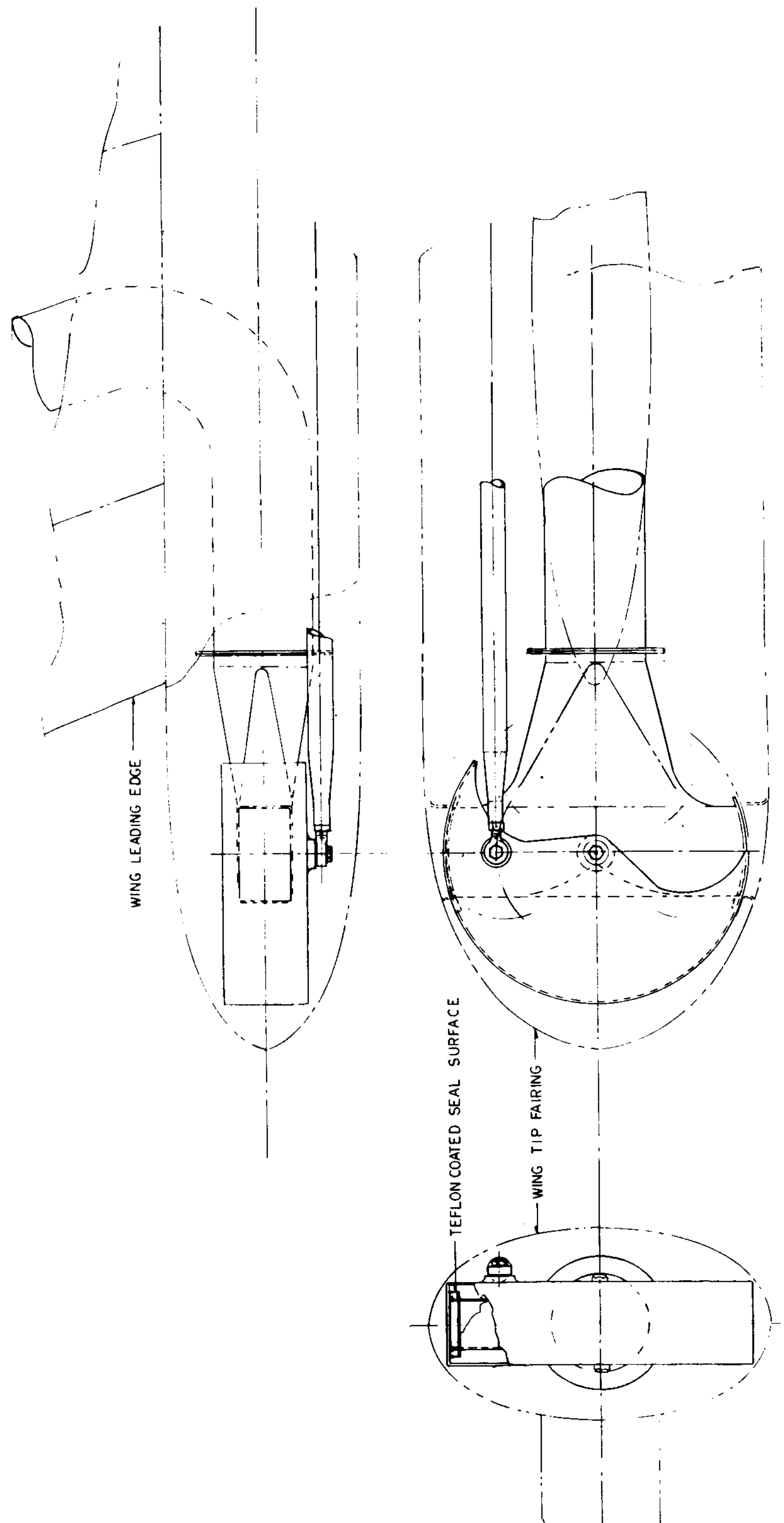


FIGURE 4-2. REACTION JET NOZZLE - ROLL CONTROL

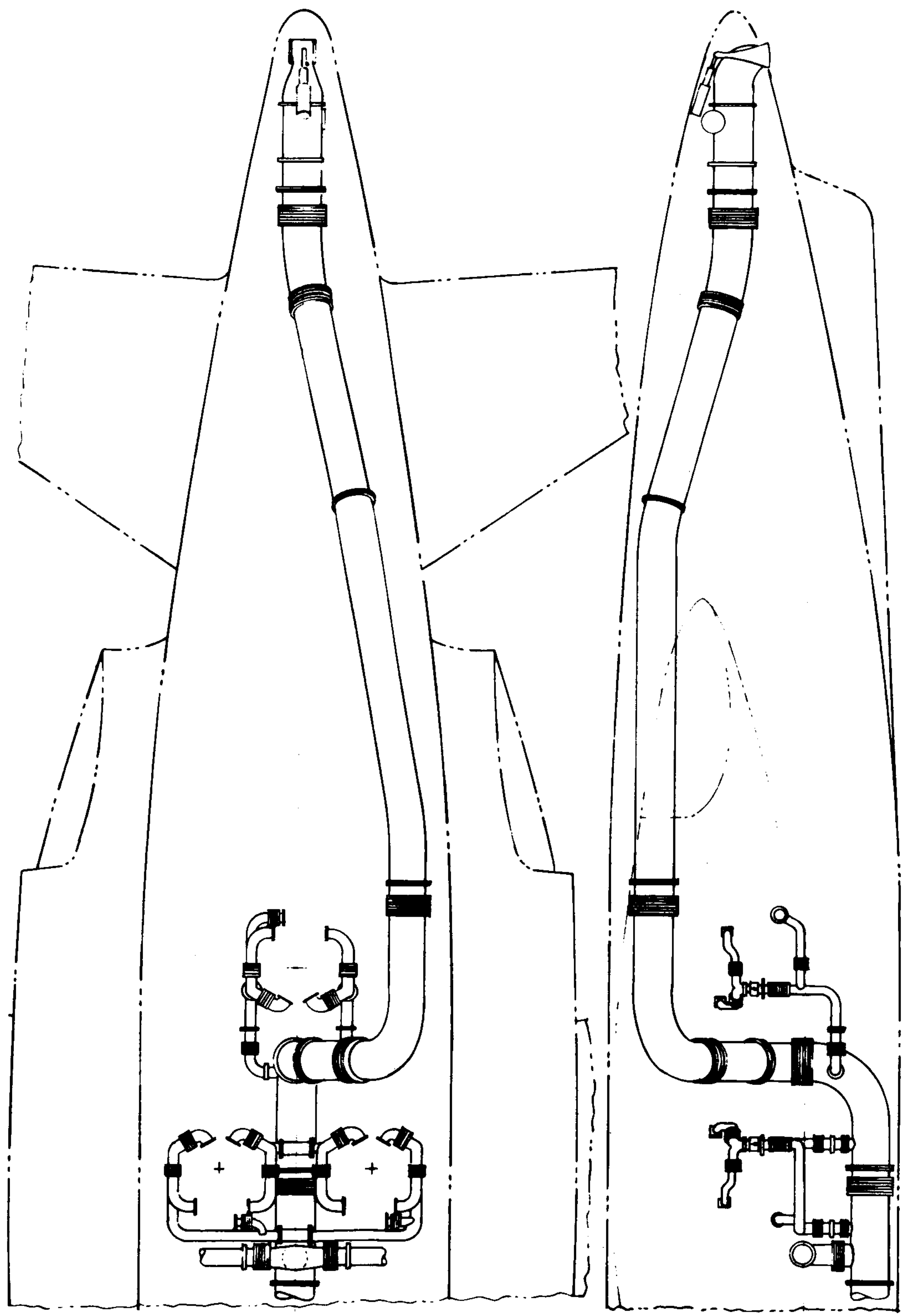


FIGURE 4-3. ALTERNATE REACTION CONTROL SYSTEM

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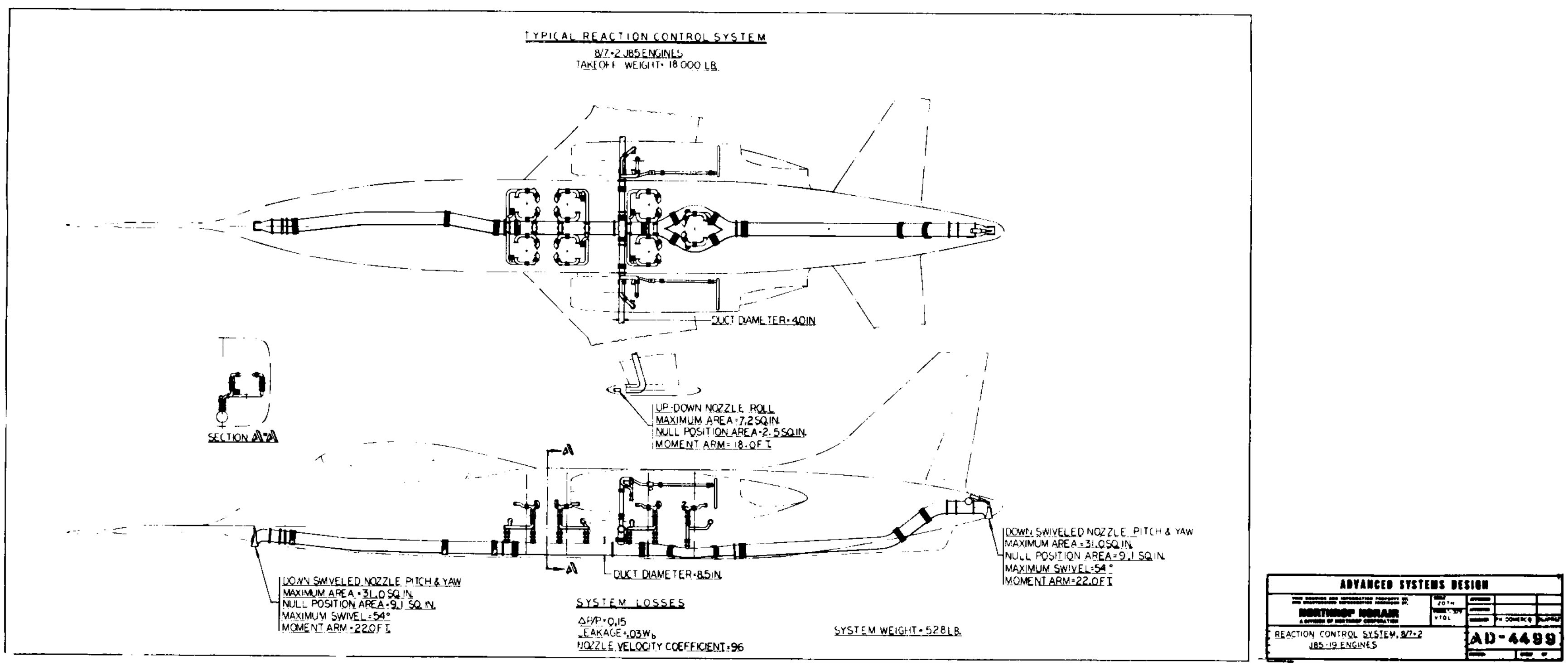

TYPICAL REACTION CONTROL SYSTEM
8/7-2 J85 ENGINES
TAKEOFF WEIGHT - 18 000 LB.
DUCT DIAMETER - 4.0 IN
SECTION A-A
UP-DOWN NOZZLE ROLL
MAXIMUM AREA - 7.2 SQ IN.
NULL POSITION AREA - 2.5 SQ IN.
MOMENT ARM - 18.0 FT.
DOWN SWIVELED NOZZLE PITCH & YAW
MAXIMUM AREA - 31.0 SQ IN.
NULL POSITION AREA - 9.1 SQ IN.
MAXIMUM SWIVEL - 54°
MOMENT ARM - 22.0 FT.
DUCT DIAMETER - 8.5 IN.
SYSTEM LOSSES
ΔP/P - 0.15
LEAKAGE - .03 W
NOZZLE VELOCITY COEFFICIENT - .96
SYSTEM WEIGHT - 528 LB.
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
REACTION CONTROL SYSTEM, 8/7-2
J85-19 ENGINES
AD-4499

FIGURE 4-3A

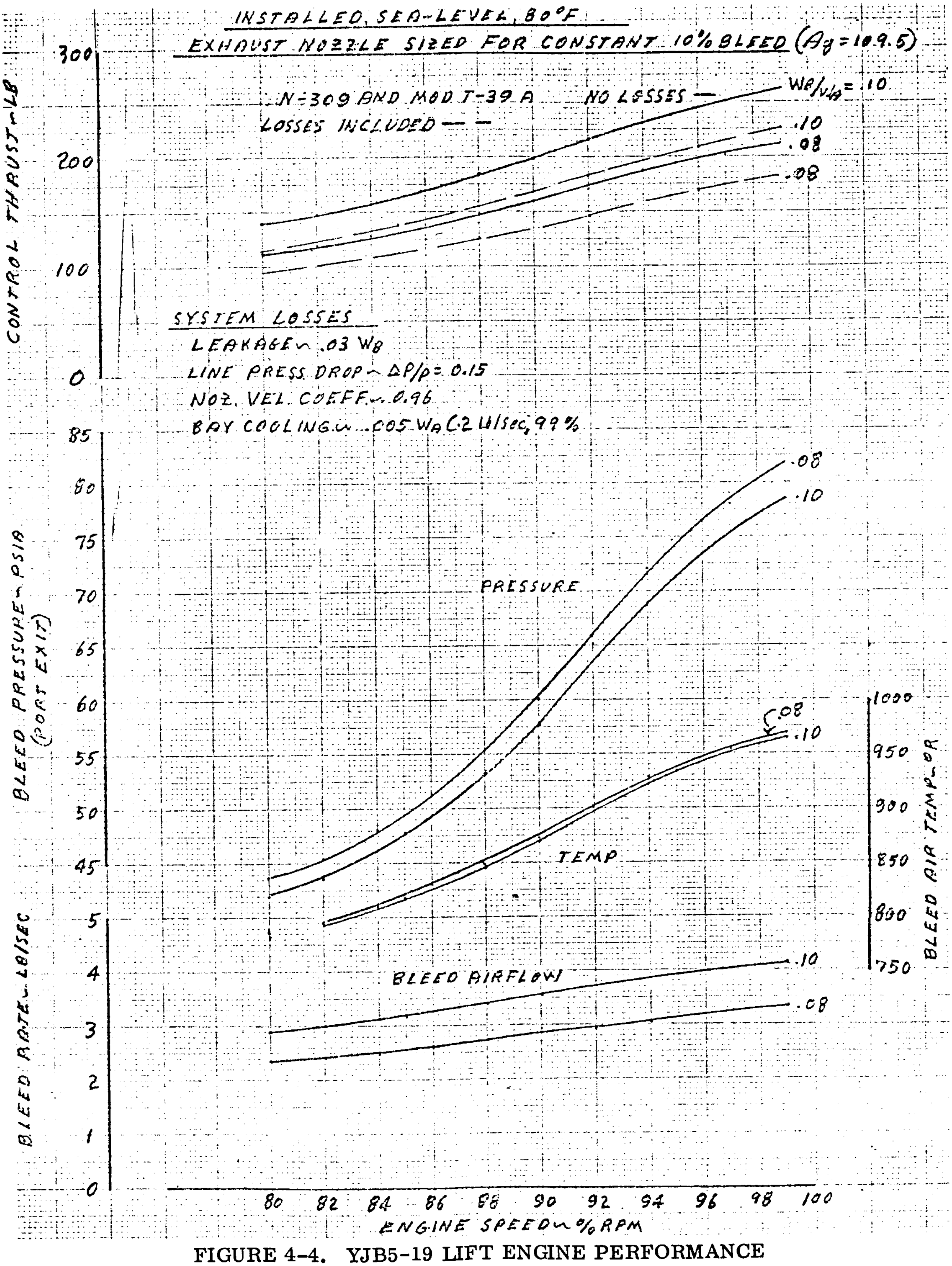


FIGURE 4-4. YJB5-19 LIFT ENGINE PERFORMANCE
BLEED AIR QUALITY AND CONTROL THRUST

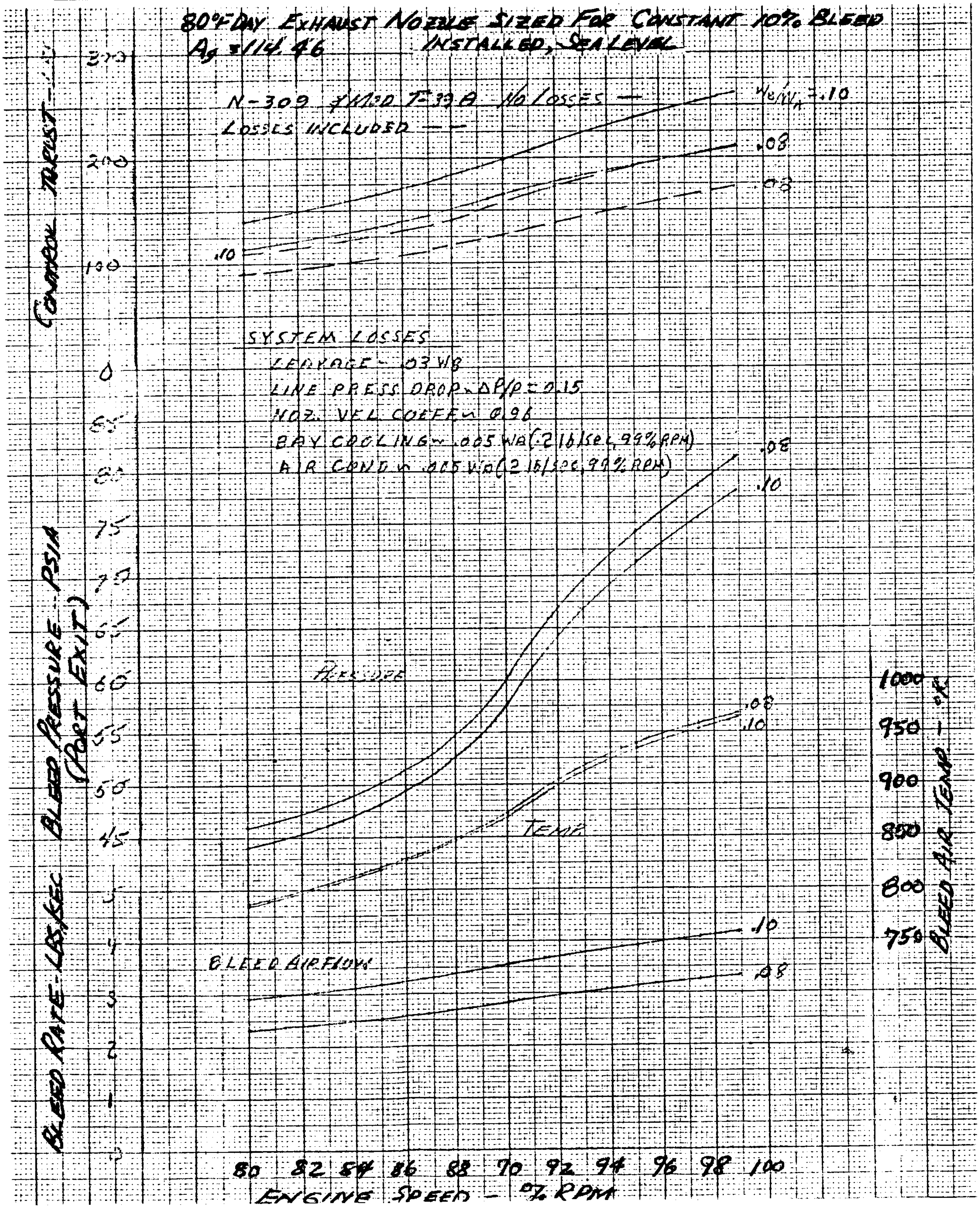


FIGURE 4-5. J85-19 L/C ENGINE PERFORMANCE - BLEED AIR QUALITY AND CONTROL THRUST

reaction nozzles is 0.95; the discharge coefficient does not affect the available control thrust since the discharge area is variable.

With the available control thrust per engine established, the analysis was directed to determining the total control thrust and moments available from each configuration. To obtain a "working" plot depicting available control thrust as a function of engine speed, bleed rate, and total nozzle area, engine bleed characteristics were matched with flow continuity to obtain Figure 4-6 and, in turn, Figure 4-7. The latter figure defines the control thrust available from a nozzle of specified area as a function of total nozzle area and engine speed. These data are applicable to either 8, 9 or 10-engine installations, providing the correct abscissa is used as noted on the plot. Line pressure drops were varied as a function of bleed rate to approach more closely actual losses: 15, 12, and 10 percent pressure drops for bleed rates of 10, 8, and 6 percent, respectively. However, for the cases where 100 percent of the control requirement on a single axis is specified, a line pressure drop of 20 percent was used. These pressure drops were substantiated for the N-309 duct layout, but are applicable to the Mod. T-39A design. The data of Figure 4-7 provided the means to establish the adequacy of any selected nozzle schedule in meeting specified control requirements.

Although the requirement for 100 percent control about each axis separately during light-weight hover was not critical, it sized the maximum and "null" nozzle areas and dictated the mechanization of the control system. The lift-to-weight ratio for the light-weight hover condition is 1.09 for the N-309 and 1.11 for the Mod. T-39A based on empty weight. This allows a fuel margin equal to 5 percent of the empty weight and out-of-ground interference lift losses of 4 percent and 6 percent for the N-309 and Mod. T-39A, respectively. The control requirement of 60 percent control about each axis simultaneously during light-weight hover with all engines operating proved critical due to reduced control thrust at the low engine speeds for hover at light weight. The other simultaneous critical control requirement was for 20 percent control in pitch and yaw and 50 percent in roll while hovering at the light-weight condition with one engine inoperative. Both of these control conditions were evaluated at maximum power to establish the ratio of lift-to-design gross weight available with these amounts of simultaneous control applied. The critical performance-control requirement was a lift/design weight ratio of 1.05 in ground effect while providing control power equal to 80 percent in pitch and 50 percent in roll and yaw. Lift losses in ground effect were 15 percent for the N-309 and 17 percent for the Mod. T-39A, which raised the required L/W ratios to 1.20 and 1.22.

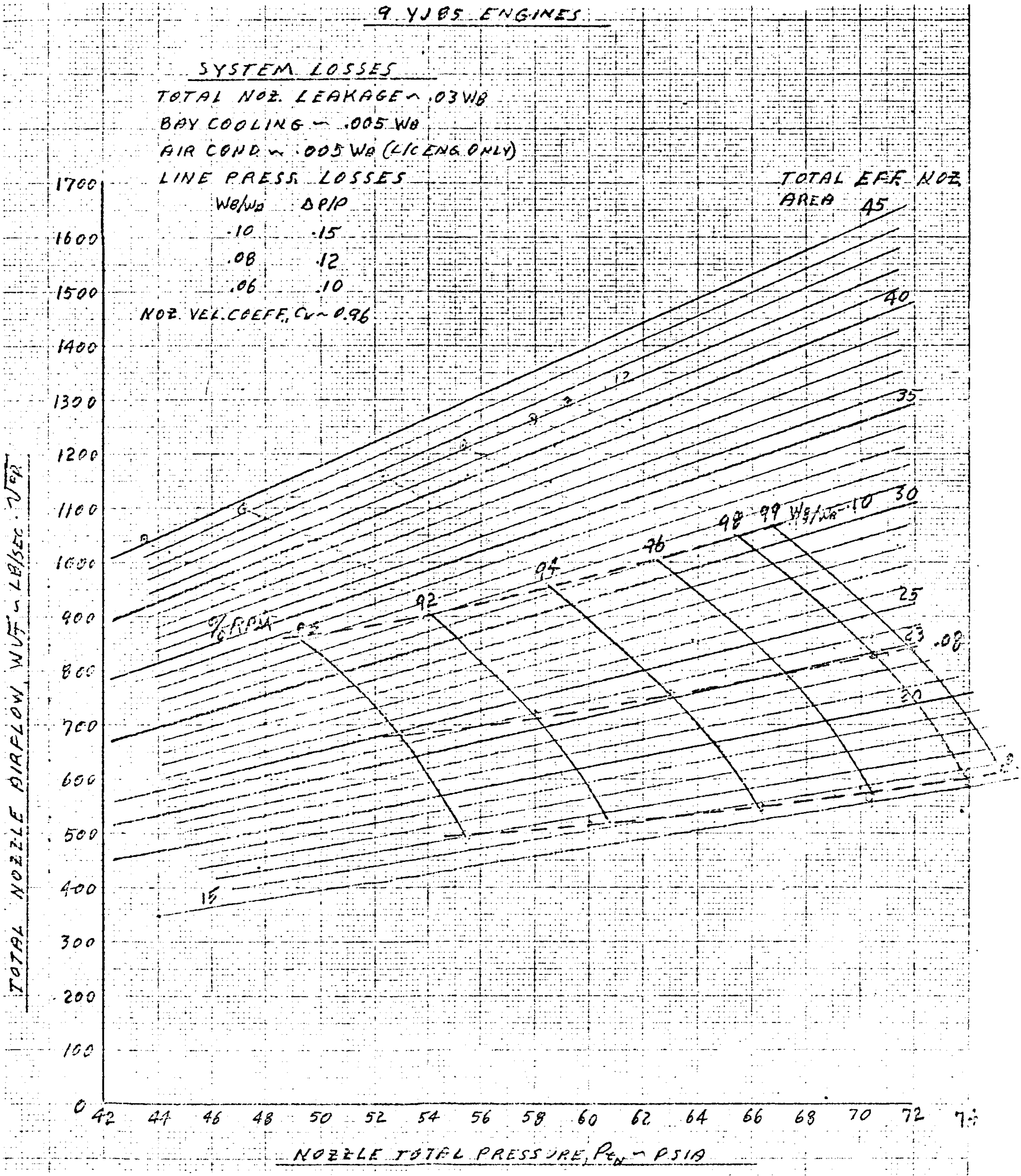


FIGURE 4-6. MATCHING OF CONTROL NOZZLE AREA WITH ENGINE COMPRESSOR BLEED CHARACTERISTICS

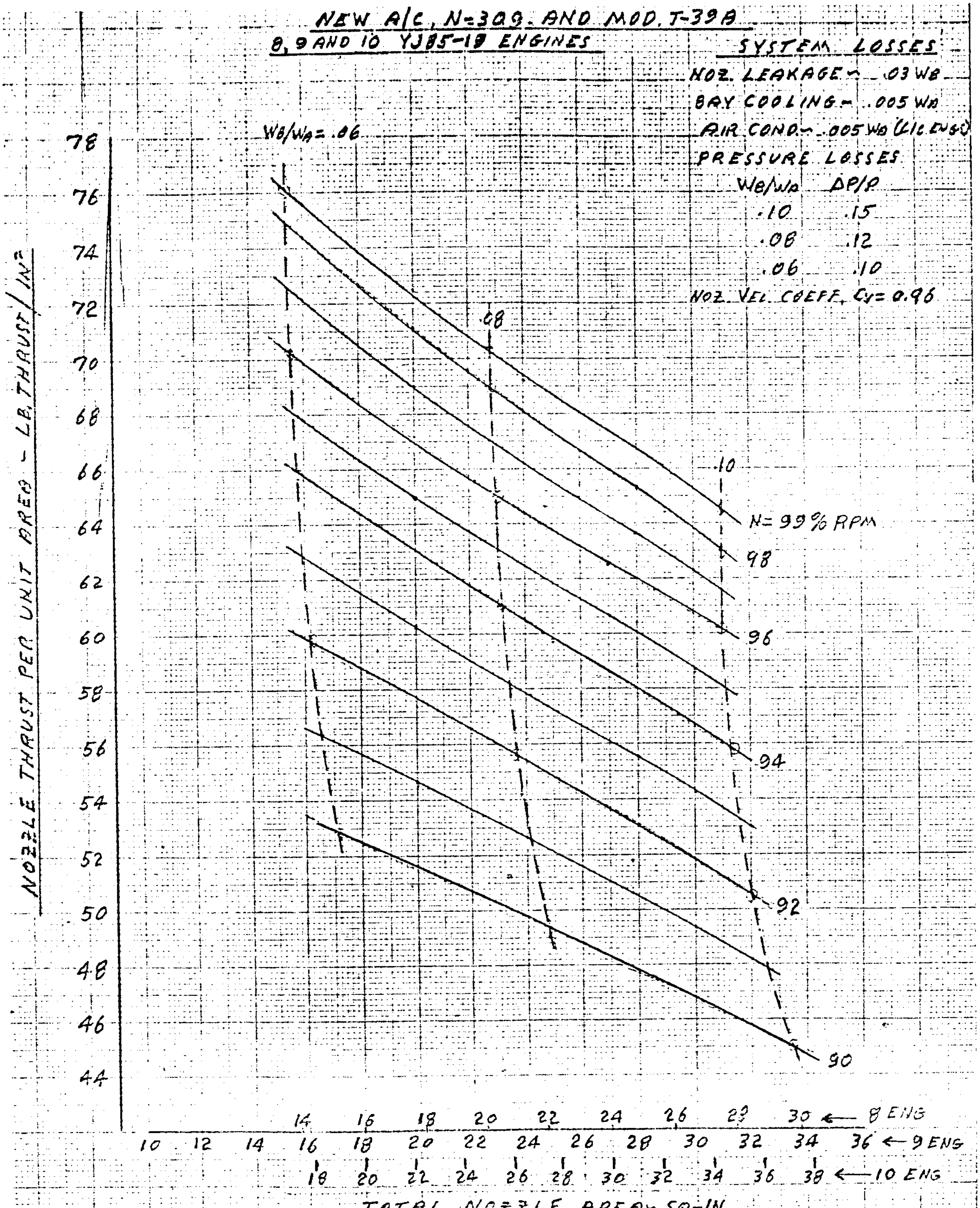


FIGURE 4-7. CONTROL THRUST PER UNIT AREA
VERSUS TOTAL NOZZLE AREA (EFFECTIVE AREAS)

All other specified control and performance-control requirements were less critical than the above.

4.3.2 N-309 (New Airplane)

Nozzle area schedules for the new aircraft design (N-309) in the composite-lift mode are shown in Figure 4-8. As indicated for the pitch/yaw nozzle, both nozzle areas increase when the nozzles are rotated beyond 40 degrees. The reason for this is to limit the amount of rotation required to obtain the required maximum yaw moment and still retain "null" areas sufficiently small so as to allow the attainment of maximum required roll moments without forcing a reduction in pitch nozzle areas. This also applies to the "null" areas of the roll nozzles which were sized to allow maximum pitch control with no change in roll nozzle areas. Each roll nozzle is scheduled to open an additional 2.2 sq.-in. after the opposite nozzle closes and before it starts to discharge in the up direction. This presents no problem in the mechanical design of the nozzle area-control system, but allows the attainment of 50 percent of the available roll control with minimum cross coupling into the pitch axis and lift.

The total nozzle area with all nozzles at their respective null areas for the zero control demand case is 23.3 sq.-in. The resulting bleed rate varies only slightly from 8.0 percent in the engine speed range of interest between 92 and 99 percent rpm. An effective total nozzle area of 32.0 sq.-in. results in a bleed rate of 10.0 percent, the nominal limit bleed rate of the YJ85-19 engine.

Based on the nozzle schedules (Figure 4-8) and the data of Figure 4-7, plots of available control moments for engine speeds of 92 and 99 percent RPM were constructed and are shown in Figures 4-9 and 4-10 for pitch and yaw and in Figure 4-11 for roll. These data show that the control system provides more than 100 percent of the lightweight control requirements for separate axes at an engine speed of 92 percent RPM. With reference to Figure 3-18 (Section 3) an engine speed of 91 percent RPM is required to hover with 100 percent pitch control, since all the control thrust is contributing to lift. The engine speed required to hover with maximum yaw and roll control applied separately is about 91.5 percent RPM due to control system lift losses: 450 pounds for yaw and 380 pounds for roll. Figure 4-12 presents a schematic of the control system and tabulates pertinent operational information for the critical simultaneous control requirements and shows that the control requirements were met.

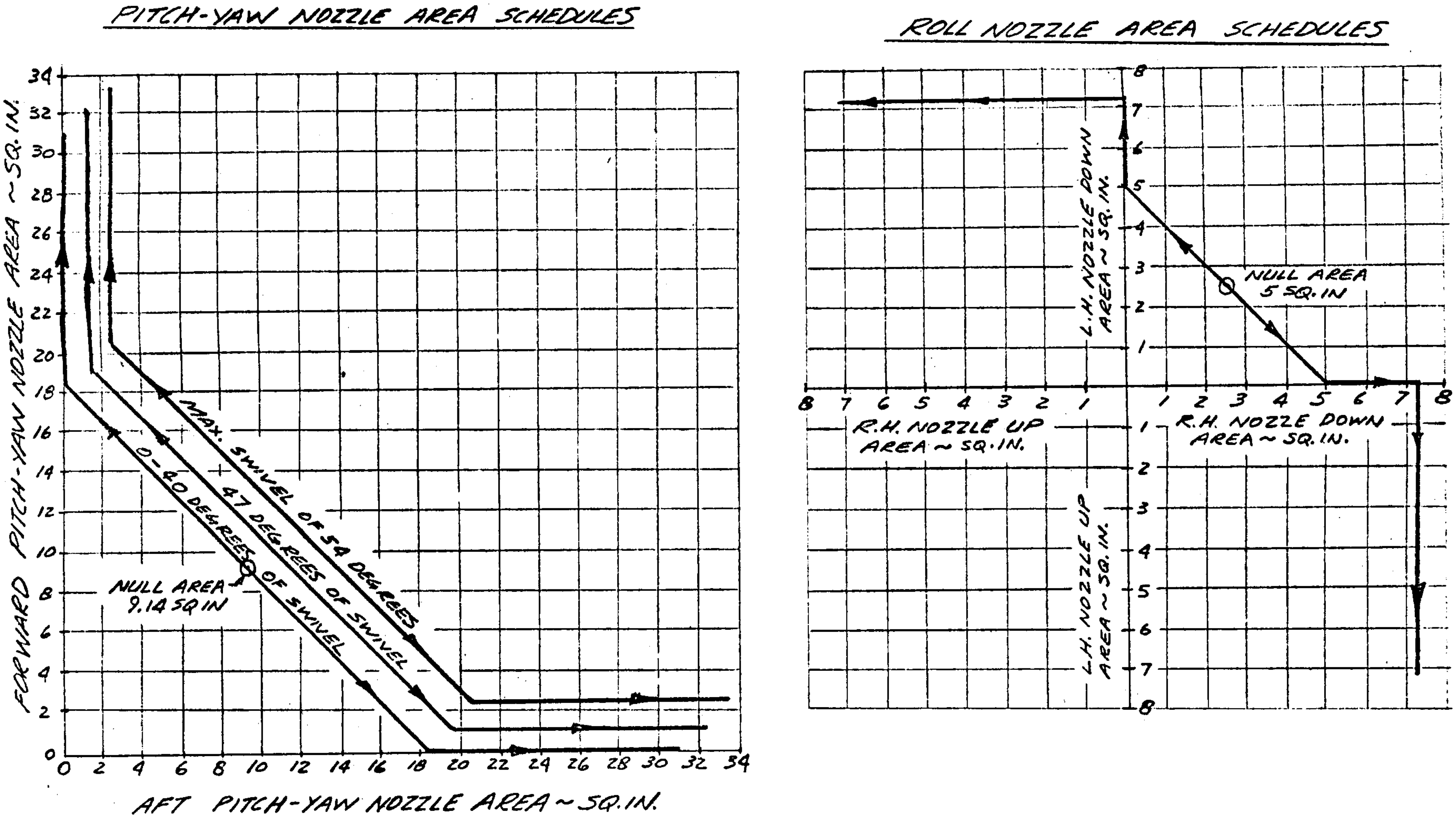


FIGURE 4-8. N-309 NOZZLE SCHEDULES

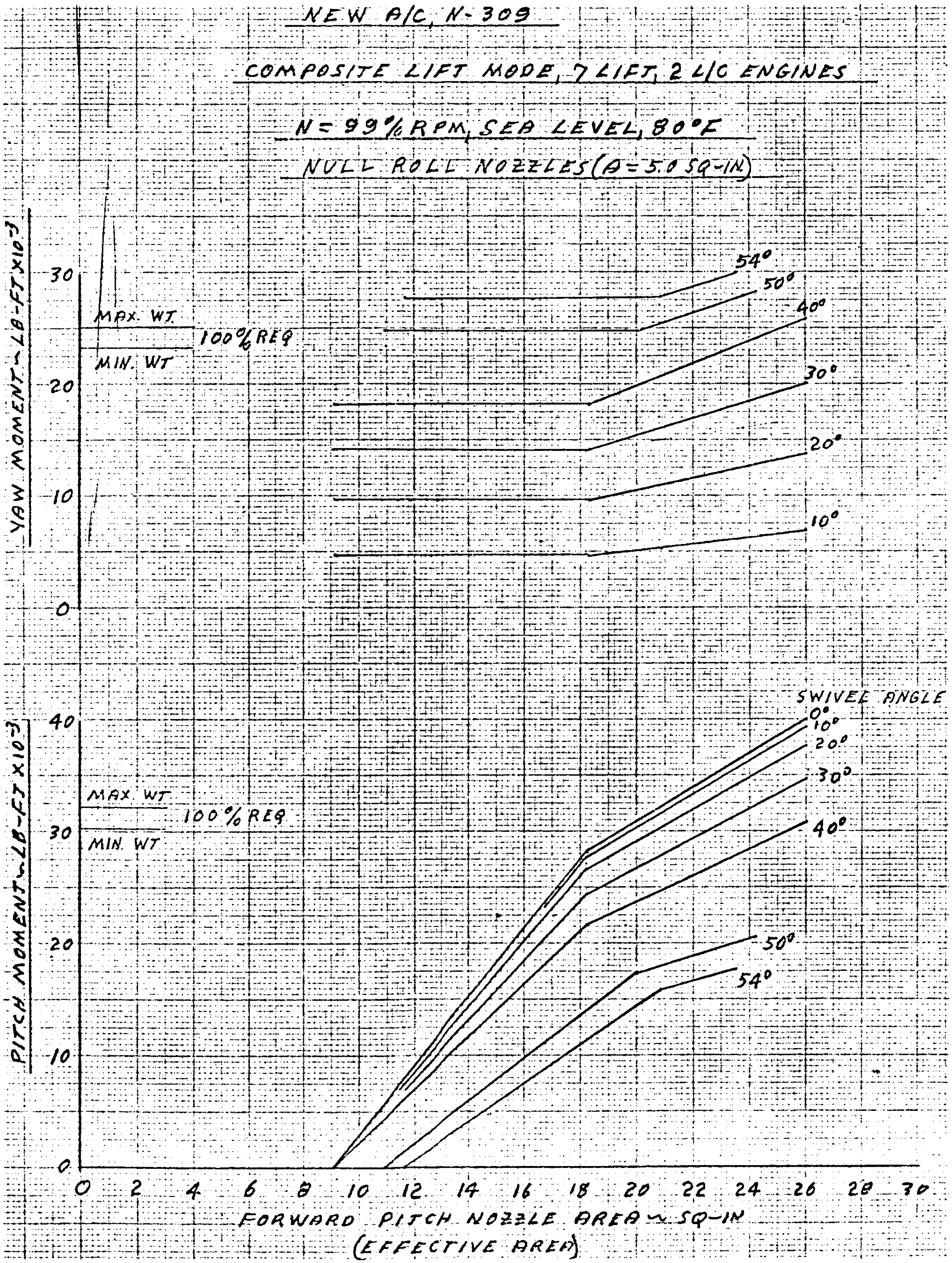


FIGURE 4-9. AVAILABLE PITCH AND YAW CONTROL MOMENT

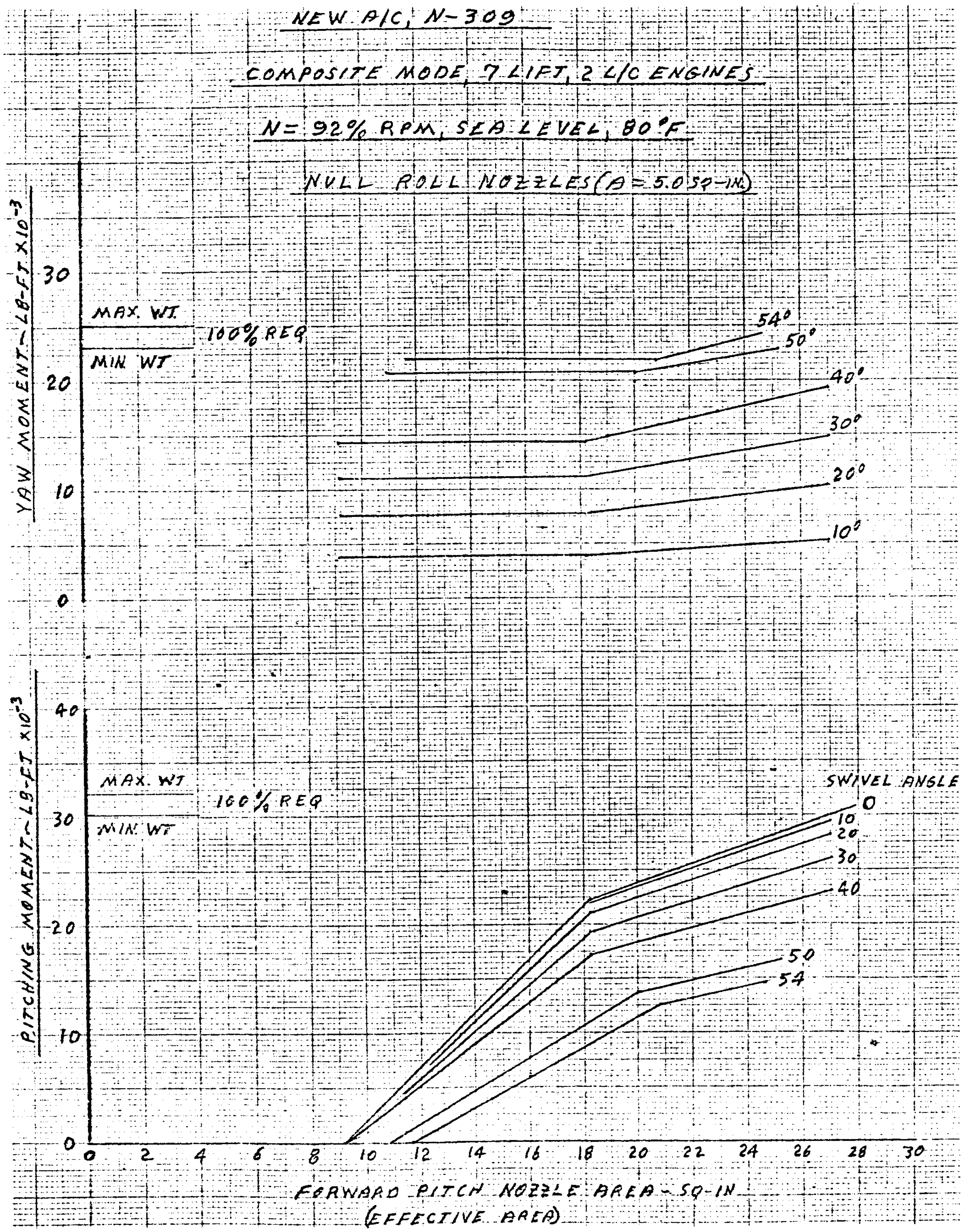


FIGURE 4-10. AVAILABLE PITCH AND YAW CONTROL MOMENT

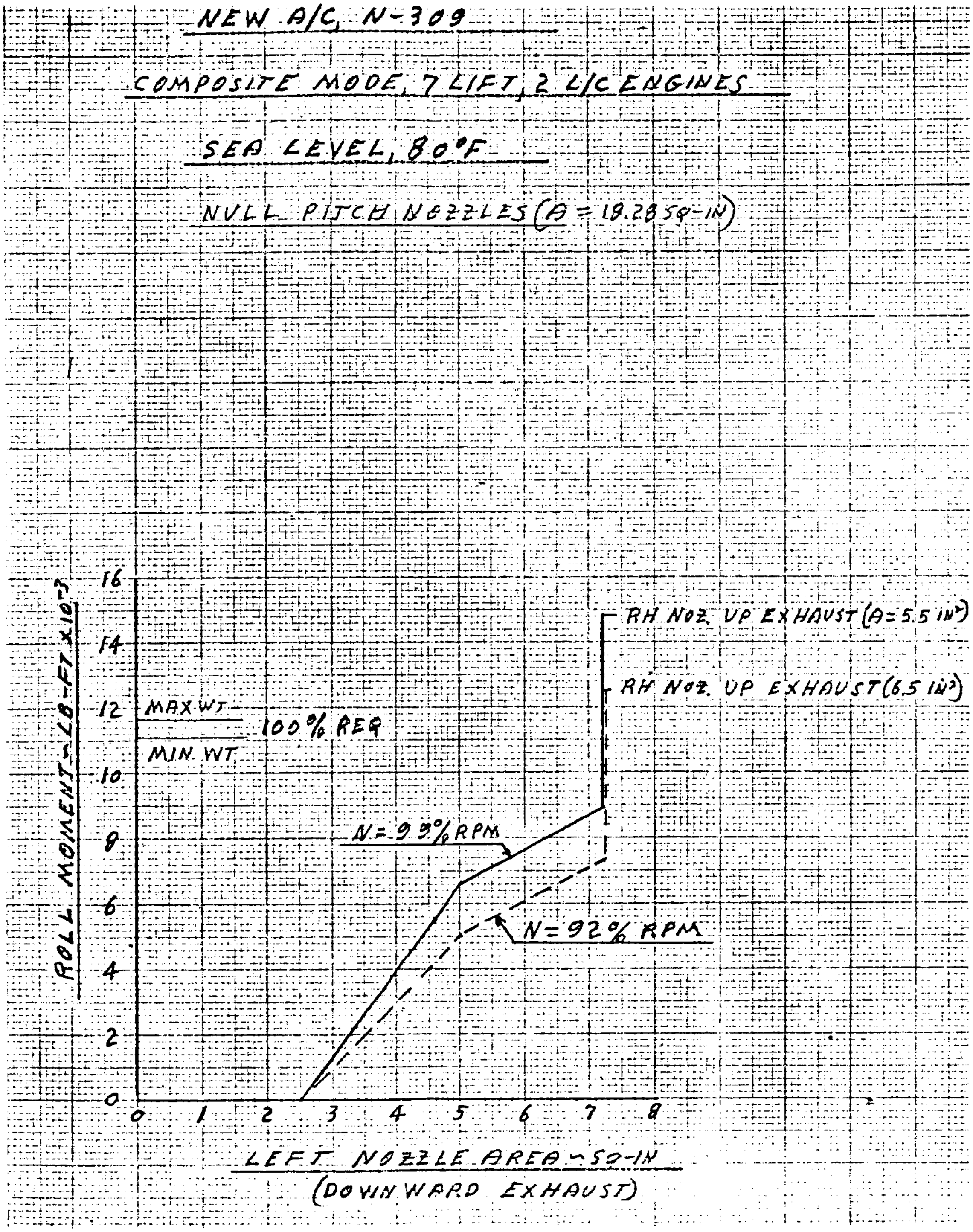
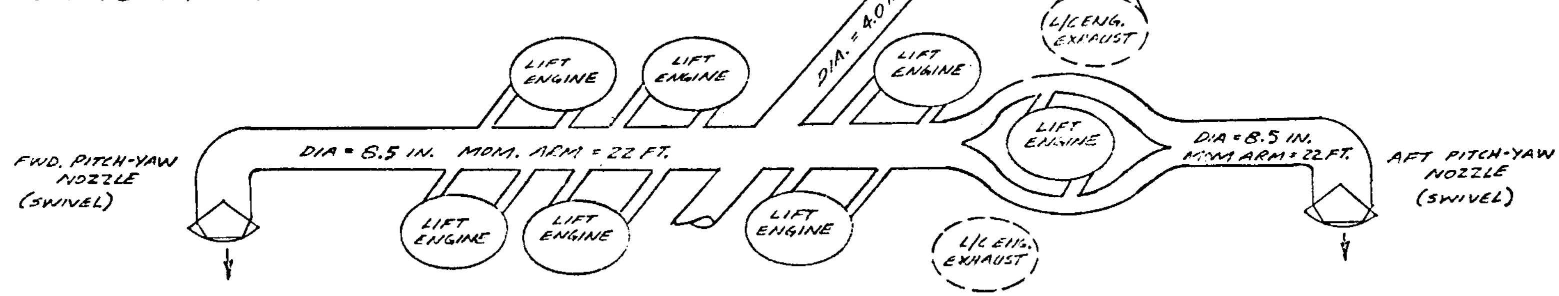


FIGURE 4-11. AVAILABLE ROLL CONTROL MOMENT

	DESCRIPTION	CONTROL THRUST (LB) PITCH	ROLL	YAW	ENGINE LIFT (LBS)	CONTROL LIFT (LBS)	TOTAL LIFT (LB)	L/W	BLEED (%)
1	LIFT-OFF CONV. WT.	—	—	—	20920	1790	22710	1.26	8.5
2	MAX. PITCH MIN. WT. HOVER	1380	—	—	13,470	1638	15,108	1.10	10.0
3	MAX. ROLL MIN. WT. HOVER	—	648		14,038	1010	15,098	1.10	10.0
4	MAX. YAW MIN. WT. HOVER	—	—	1058	14,219	1060	15,279	1.10	9.3
5	60-60-60 MIN. WT. HOVER	828	389	635	14200	1178	15,378	1.10	8.5
6	60-50-50 MAX. WT.	1170	309	568	20,025	1531	21,556	1.20	8.25
7	20-50-20 MIN. WT. HOVER	716	448	211	13,958	1268	14,226	1.10	9.8 **

** FORWARD LIFT ENGINE OUT



FORWARD PITCH-YAW NOZZLE

	SWIVEL ANGLE (DEGREE)	AREA (SQ. IN.)	AIRFLOW (LB./SEC)	THRUST (LB) PITCH	YAW
1	0	9.14	12.25	700	0
2	0	0	0	0	0
3	0	9.14	9.85	505	0
4	54 LEFT	11.64	12.14	655 387	529
5	37.5 LEFT	0	0	0	0
6	25.0 LEFT	0	0	0	0
7	11.7 LEFT	15.90	17.19	889 869	180

R.H. ROLL NOZZLE

	EXHAUST DIRECTION	AREA (SQ. IN.)	AIRFLOW (LB./SEC.)	THRUST (LBS)
1	DOWN	2.51	3.40	195
2	DOWN	2.51	2.56	129
3	DOWN	7.17	6.21	324
4	DOWN	2.51	2.65	138
5	DOWN	7.17	6.35	323
6	DOWN	0.30	0.50	26
7	DOWN	7.17	6.70	347

AFT PITCH-YAW NOZZLE

	SWIVEL ANGLE (DEGREE)	AREA (SQ. IN.)	AIRFLOW (LB/SEC)	THRUST (LB) PITCH	YAW
1	0	9.14	12.25	700	0
2	0	31.00	26.70	1380	0
3	0	9.14	9.85	505	0
4	54 RIGHT	11.64	12.14	655 387	529
5	31.5 RIGHT	19.60	20.10	1040 828	635
6	25.9 RIGHT	18.30	23.20	1300 1170	568
7	11.7 RIGHT	2.40	2.95	155 153	31

L.H. ROLL NOZZLE

	EXHAUST DIRECTION	AREA (SQ. IN.)	AIRFLOW (LB/SEC)	THRUST (LBS)
1	DOWN	2.51	3.40	195
2	DOWN	2.51	2.56	129
3	UP	7.17	6.21	324
4	DOWN	2.51	2.65	138
5	UP	1.08	1.20	66
6	DOWN	4.80	6.00	335
7	UP	1.76	2.00	101

FIGURE 4-12. CONTROL SYSTEM SCHEMATIC

Another control requirement stipulated that 50 percent control about each axis shall be available when the controls are fully deflected. With the controls fully deflected the total nozzle area is about 1.4 times that required to pass an airflow equal to the maximum bleed rate of the engines. The maximum bleed rate of the YJ85-19 is internally limited to 10 percent of compressor airflow with no adverse operational effects resulting from increasing the end flow area above that just required for maximum bleed airflow. However, the nozzle total pressure decreases due to the upstream choking to a value satisfying flow continuity with a resulting nozzle thrust loss. The thrust loss trend is shown in Figure 4-13 as a function of area increase above optimum for engine power settings of 92 and 99 percent RPM. These data show only a 10 percent thrust loss for a 40 percent area increase with all engines operating. With one engine inoperative and the remainder at maximum power, the control thrust loss will not exceed 14 percent of that available at optimum areas.

In the direct-lift configuration, the required control moments are not significantly reduced although the takeoff weight is reduced to 16,300 pounds from 18,000 pounds for the composite-lift mode. With one less engine supplying bleed air, the described control system for the composite-lift mode will not provide the required 100 percent control moments during hover at light weight. Therefore, nozzle schedules were altered as shown in Figure 4-14 to obtain a system that would satisfy the requirements. In this system the roll nozzles open only on demand with one nozzle exhausting downward, and the other upwards. This reduces bleed air requirements for nominal roll moments, and eliminates cross coupling into the pitch axis, since there can be no net lift from the nozzles regardless of the applied roll moment. However, lift changes equal to the thrust of one nozzle result. Pitch nozzle "null" areas were reduced to 8.0 sq.-in. for zero pitch demand to yield a bleed rate of 6 percent for zero control demand on all axes. Yaw moment is again obtained by swiveling the pitch nozzles, but with areas of both pitch nozzles increasing in accordance with the shown schedule. The table on page 4-22 tabulates nozzle areas and control thrust provided by the system during hover at light weight.

The trends of available control thrust and lift with engine speed with control thrusts proportioned to meet the 60-60-60 and 20-50-20 control requirements at lightweight hover are shown in Figure 4-15. Figure 4-16 is a similar presentation showing the L/W ratios available and still providing 80-50-50 control with all engines and 20-50-20 control with one engine inoperative. These data show that all requirements were met.

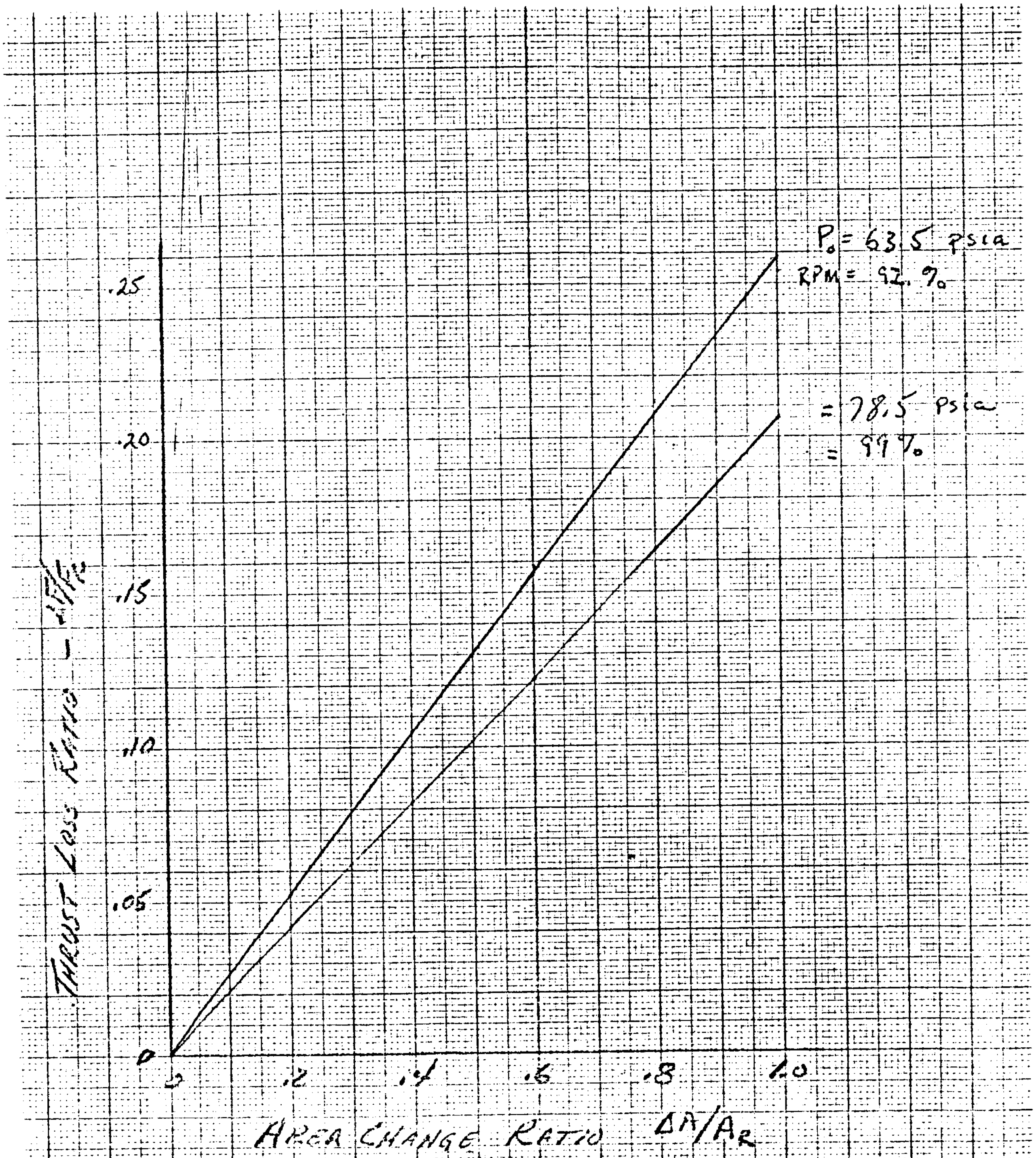


FIGURE 4-13. THRUST LOSS DUE TO UP STREAM CHOCKING

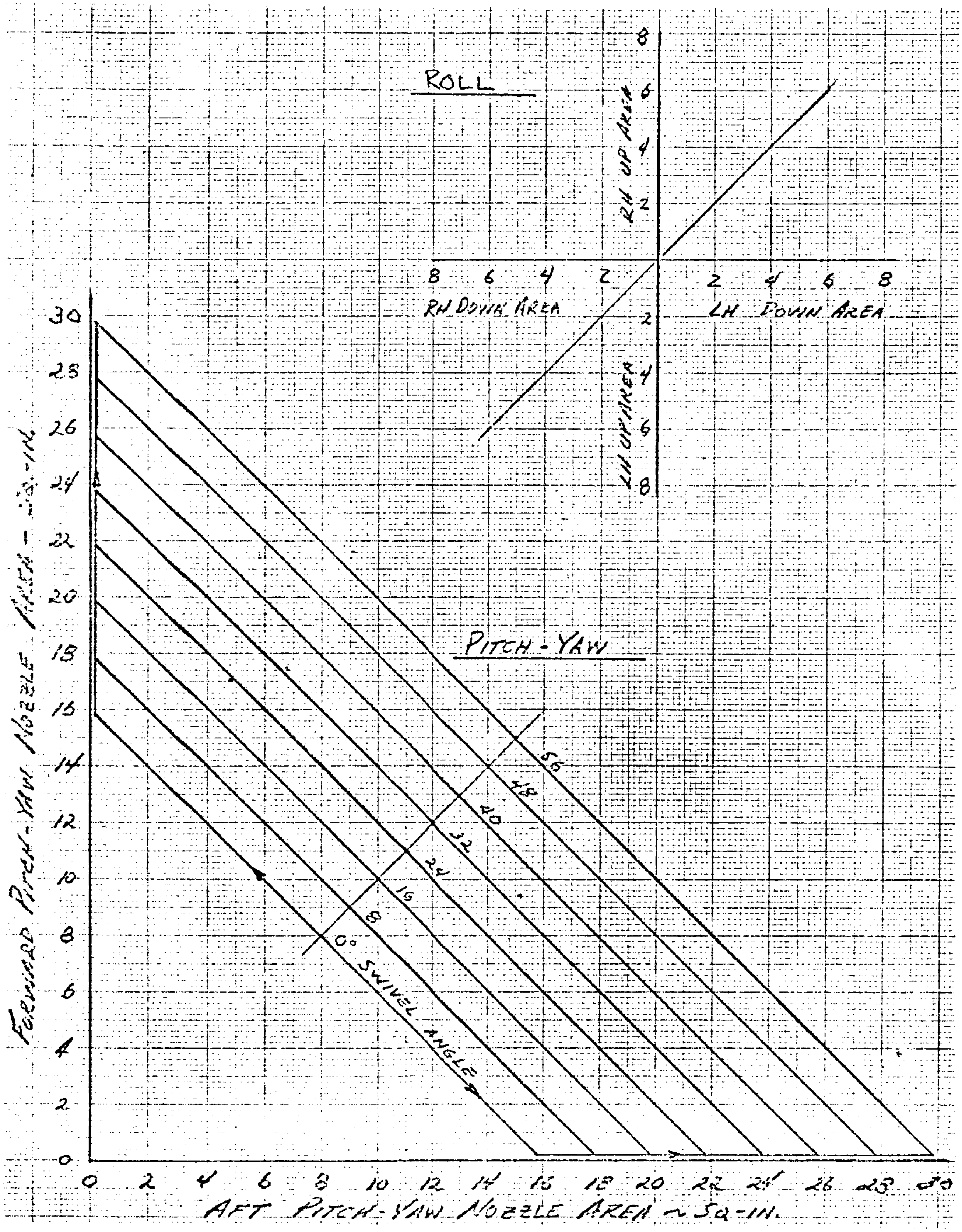


FIGURE 4-14. N-309 DIRECT-LIFT MODE NOZZLE SCHEDULES

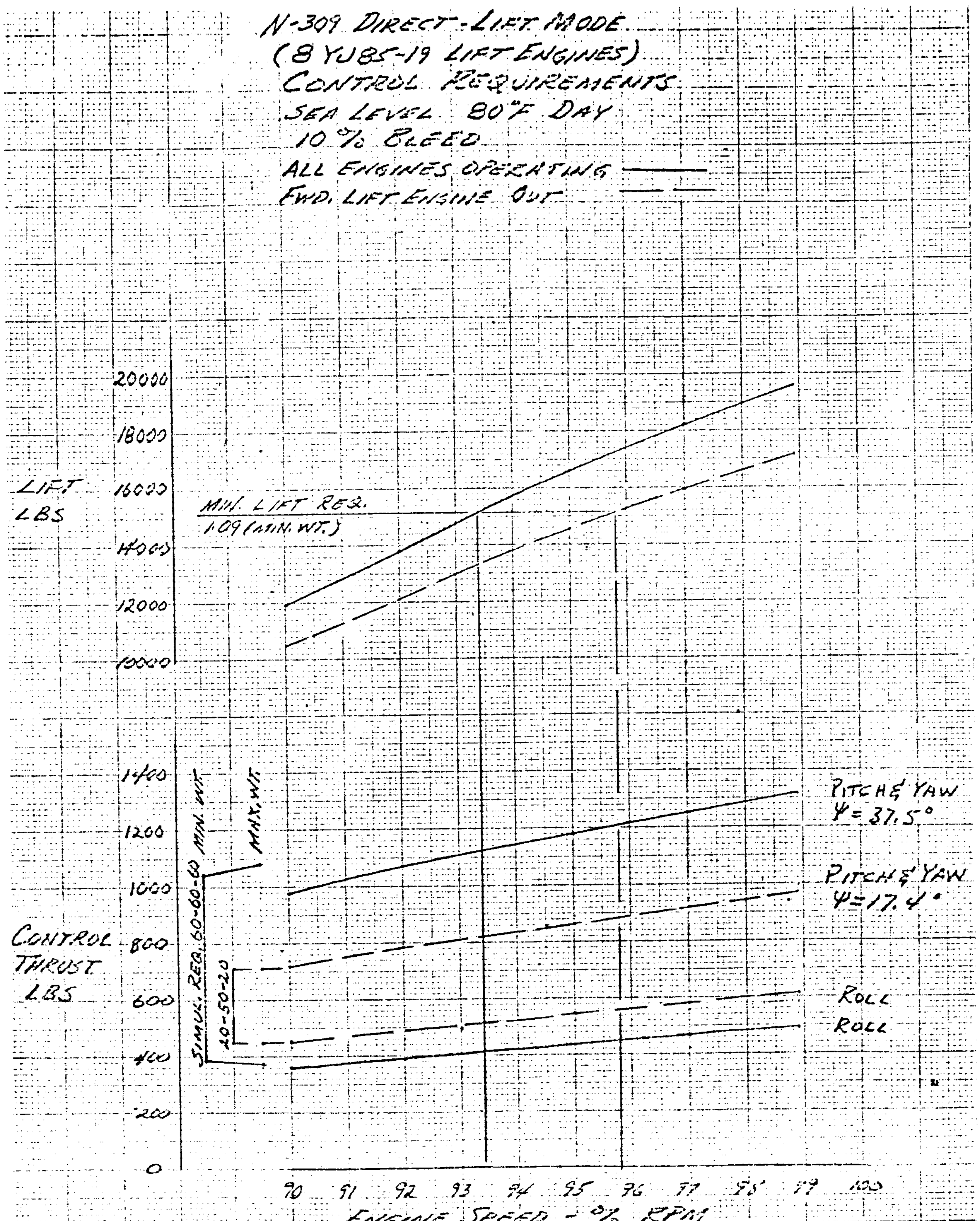


FIGURE 4-15. AVAILABLE CONTROL THRUST AND LIFT VERSUS ENGINE SPEED

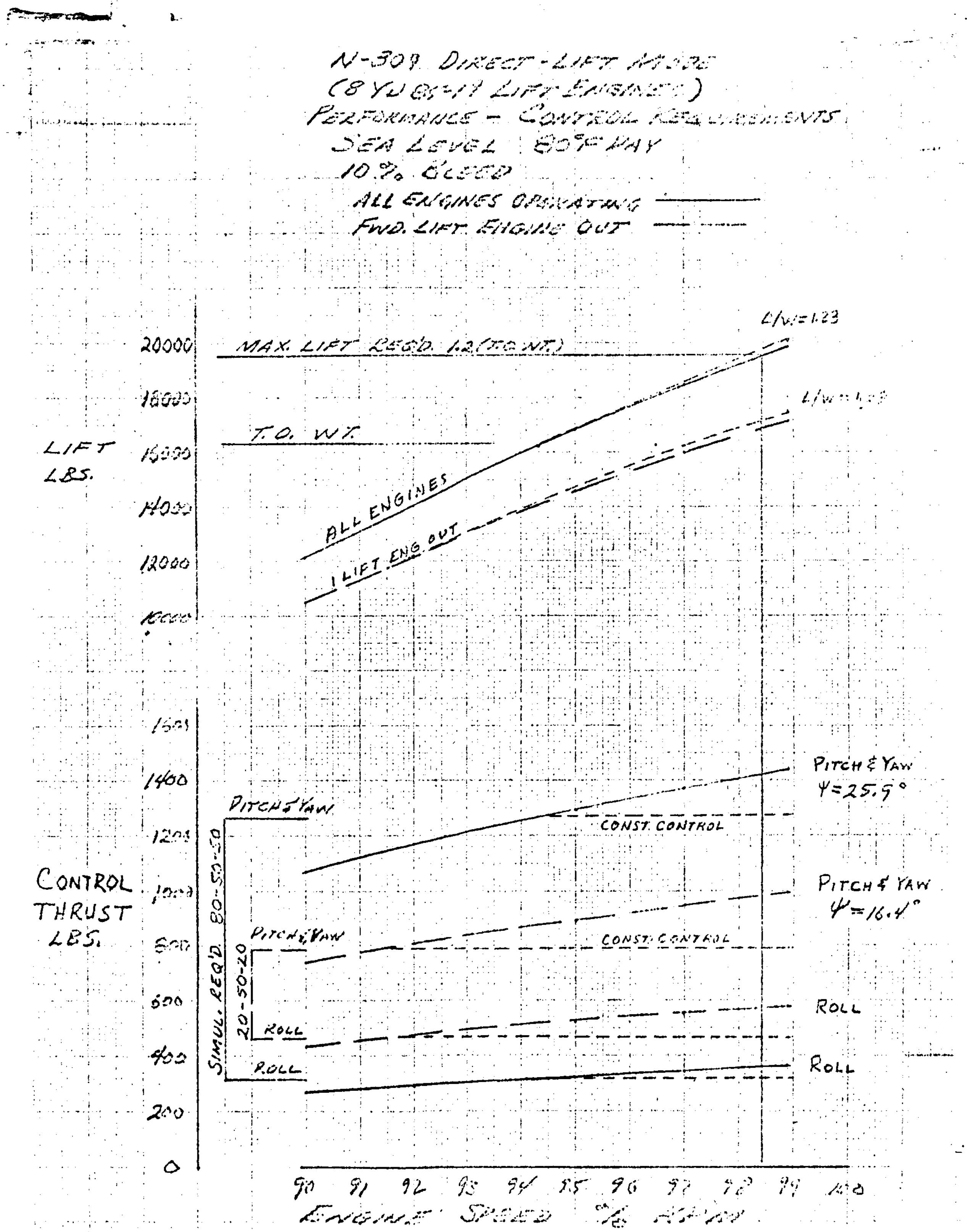


FIGURE 4-16. AVAILABLE CONTROL THRUST AND LIFT VERSUS ENGINE SPEED

N-309, Direct-Lift Mode
100 Percent Individual Axis Control
Light-Weight Hover
Pitch Mom-Arm ~ 22.0 ft. Roll Mom-Arm ~ 18.0 ft.

Req. Control Moment ~ lb-ft	Fwd. Noz. A ~ in^2	Fwd. Noz. Thrust ~ lb	Aft. Noz. A ~ in^2	Aft. Noz. Thrust ~ lb	Swivel Angle	WB/WA	L/W	RPM
Pitch ~ 30,445	25.0	1385	0	0	0	.091	1.09	92.7
Yaw ~ 23,328	13.5	745	13.5	745	47	.097	1.09	93.2

	L.H. Noz. A ~ in^2	L.H. Noz. Thrust ~ lb	R.H. Noz. A ~ in^2	R.H. Noz. Thrust ~ lb	WB/WA	L/W	RPM
Roll ~ 11,627	6.1 dn	326	6.1 up	326	.10	1.09	93.0

The different nozzle schedules indicate that the nozzle control system, if not the nozzles themselves, would have to be changed when the N-309 was converted to the direct-lift mode of operation. However, the nozzle schedules shown for the direct-lift configuration would also be suitable for the composite-lift mode with only minor consequences to the lift capability.

4.3.3 Mod. T-39A

Either of the two nozzle schedules shown for the N-309 could be used for the Mod. T-39A airplane with suitable increases in the nozzle areas to accommodate bleed air from ten engines. The N-309, direct-lift mode, system was used as a model to demonstrate adequate control power for Mod. T-39A, with nozzle schedules as shown in Figure 4-17.

System operation for the light-weight hover case at L/W = 1.11 with 100 percent control power about each axis separately is shown in the table on page 4-26.

Control and lift available as a function of engine speed for the simultaneous control requirements of 60-60-60 with all engine operating and 20-50-20 with one L/C engine inoperative are shown in Figure 4-18. The performance-control test case of a L/W of 1.22 and a simultaneous 80-50-50 control requirement is shown in Figure 4-19. For this case the interference lift loss of 17 percent is added to the basic lift margin of 5 percent.

As indicated, the available control power and lift met the requirements.

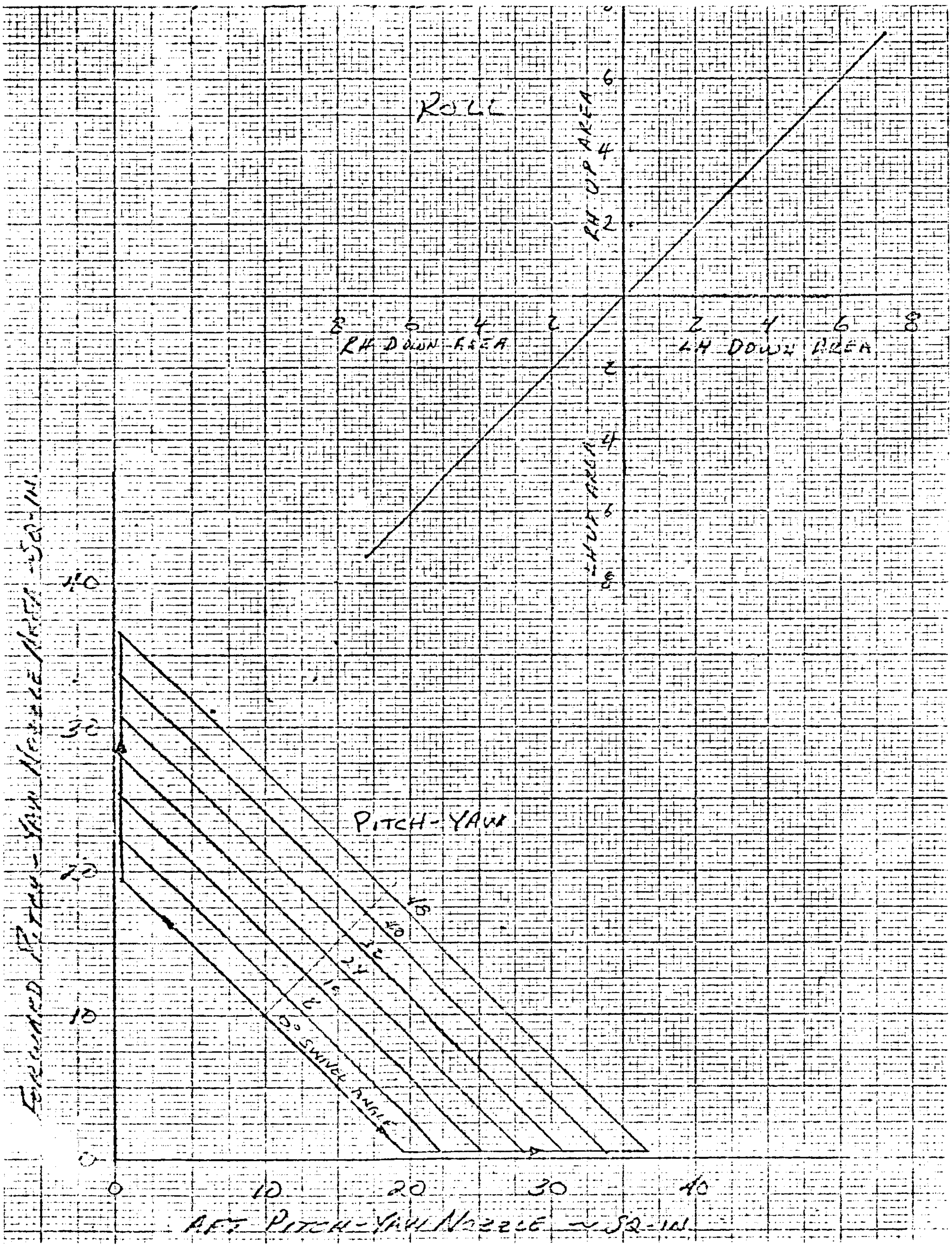


FIGURE 4-17. MODE T-39A COMPOSITE-LIFT MODE NOZZLE SCHEDULES

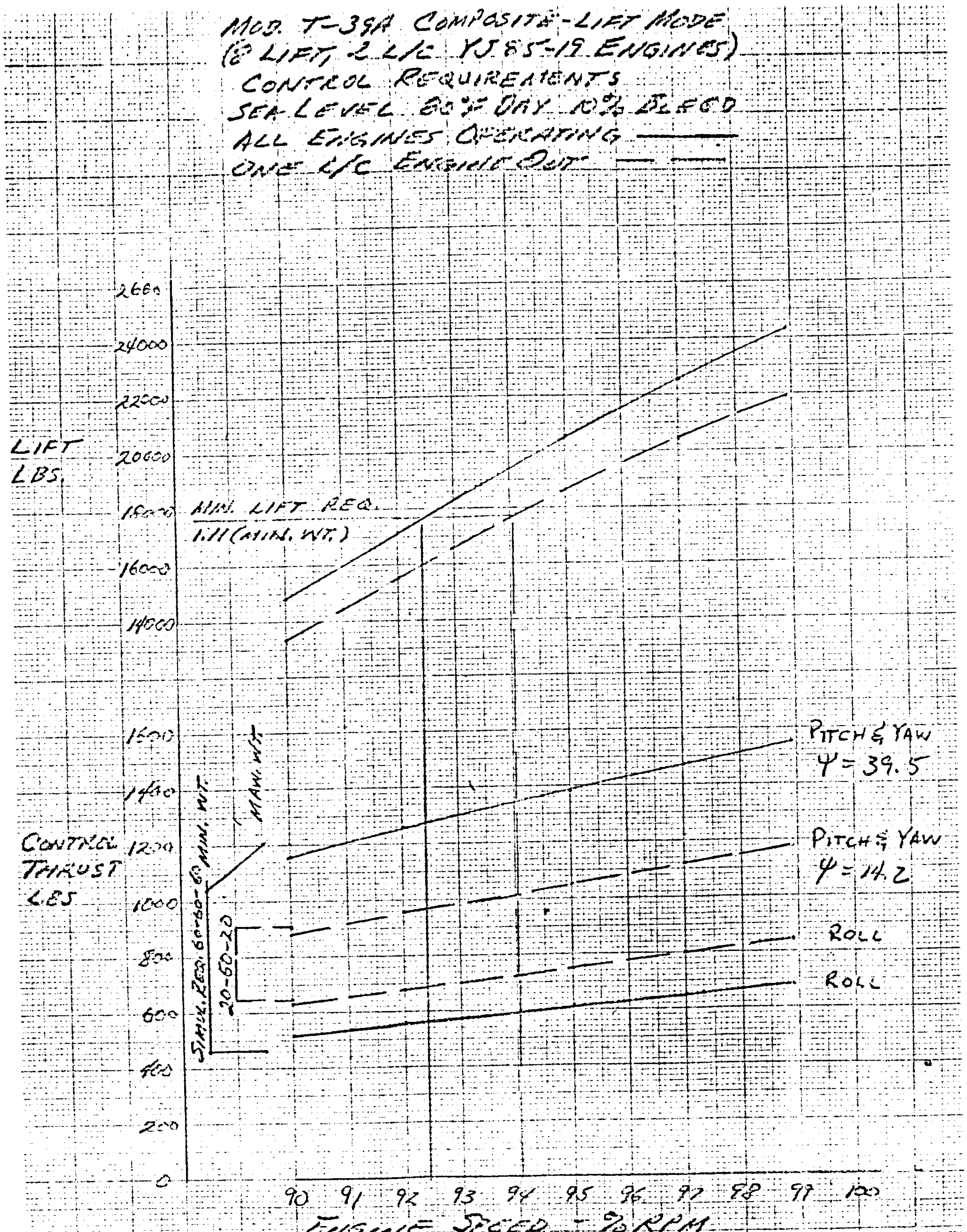


FIGURE 4-18. AVAILABLE CONTROL THRUST AND LIFT VERSUS ENGINE SPEED

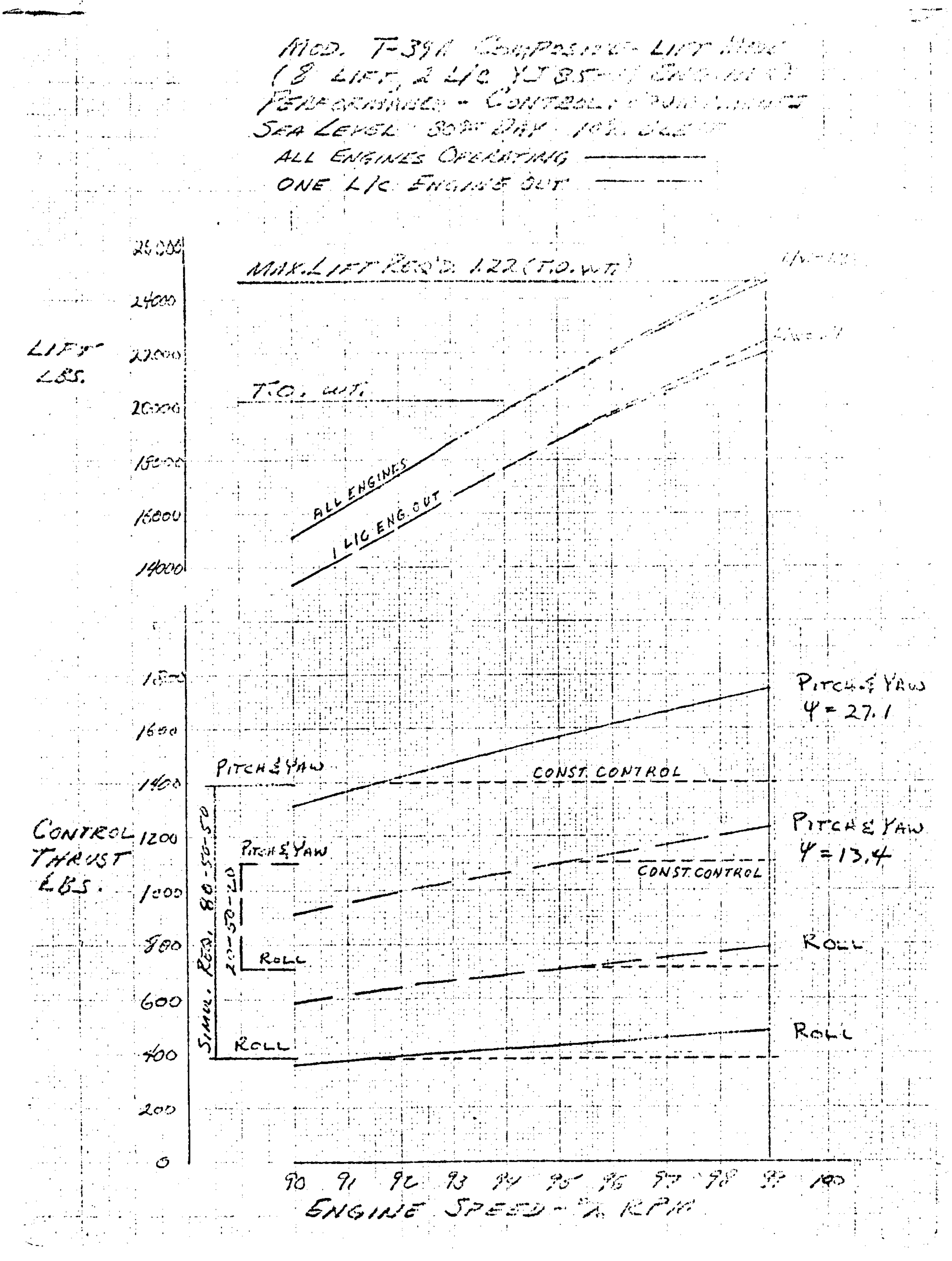


FIGURE 4-19. AVAILABLE CONTROL THRUST AND LIFT VERSUS ENGINE SPEED

Mod. T-39A Composite-Lift Mode
100 Percent Individual Axis Control
Light Weight Hover
Pitch Mom-Arm ~ 22.3 ft. Roll Mom Arm ~ 23.2 ft.

Req. Control Moment ~ lb-ft	Fwd. Noz. A ~ in^2	Fwd. Noz. Thrust ~ lb	Aft. Noz. A ~ in^2	Aft. Noz. Thrust ~ lb	Swivel Angle	WB/WA	L/W	RPM
Pitch ~ 29,700	25.0	1360	0	0	0	.078	1.11	91.8
Yaw ~ 24,900	16.8	900	16.8	900	38	.095	1.11	92.3
	L.H. Noz. A ~ in^2	L.H. Noz. Thrust ~ lb	R.H. Noz. A ~ in^2	R.H. Noz. Thrust ~ lb	WB/WA	L/W	RPM	
Roll ~ 17,900	7.5 up	387	7.5 dn	387	.098	1.11	92.1	

4.4 CROSS COUPLING

As previously stated, the objective of the control system design was to minimize control cross coupling into the remaining axes and changes in lift on the application of a control moment.

The application of a pitching moment produces insignificant lift changes and no roll moment, since the sum of engine and control thrust remains essentially the same for small changes in bleed rate.

In the N-309 design, the roll nozzles are 30.0 inches aft of the center of pitch rotation. Therefore, any change in the net lift of the roll nozzles induces a pitching moment. Induced pitching moment and lift resulting from the application of roll control during hover are shown in Figure 4-20. No pitch or lift changes are induced up to 50 percent of the maximum roll requirement and only changes of 300 lb. in lift and 700 lb-ft in pitch for a maximum applied roll moment.

The application of yaw control introduces no roll moment providing the line-of-reaction of the pitch nozzle intersects the roll axis, which is assumed for the current designs. As shown by Figure 4-21 the application of only a yaw moment causes only a lift loss due to angularity of the pitch-yaw nozzle to the vertical. The lift loss is 300 lb for a 100 percent yaw moment. For the simultaneous application of pitch and yaw, a side force of 620 lb results in addition to the lift loss, as also shown in Figure 4-21 for the simultaneous 60-60-60 control case. With a pitch moment of 60 percent of the maximum established, the application of a 60 percent yaw moment induces an

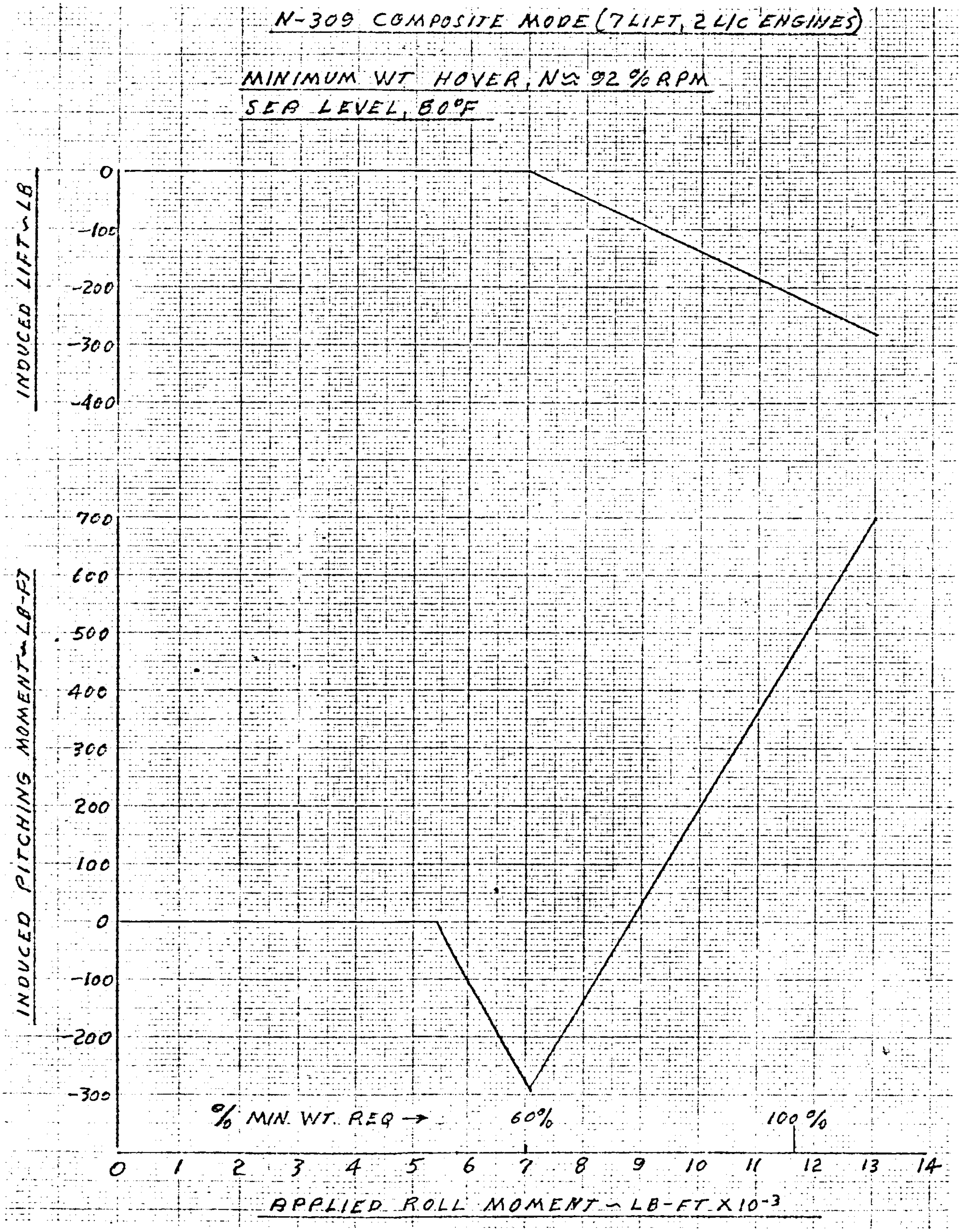


FIGURE 4-20. PITCH AND LIFT CROSS-COUPLING RESULTING FROM APPLICATION OF ROLL CONTROL DURING HOVER

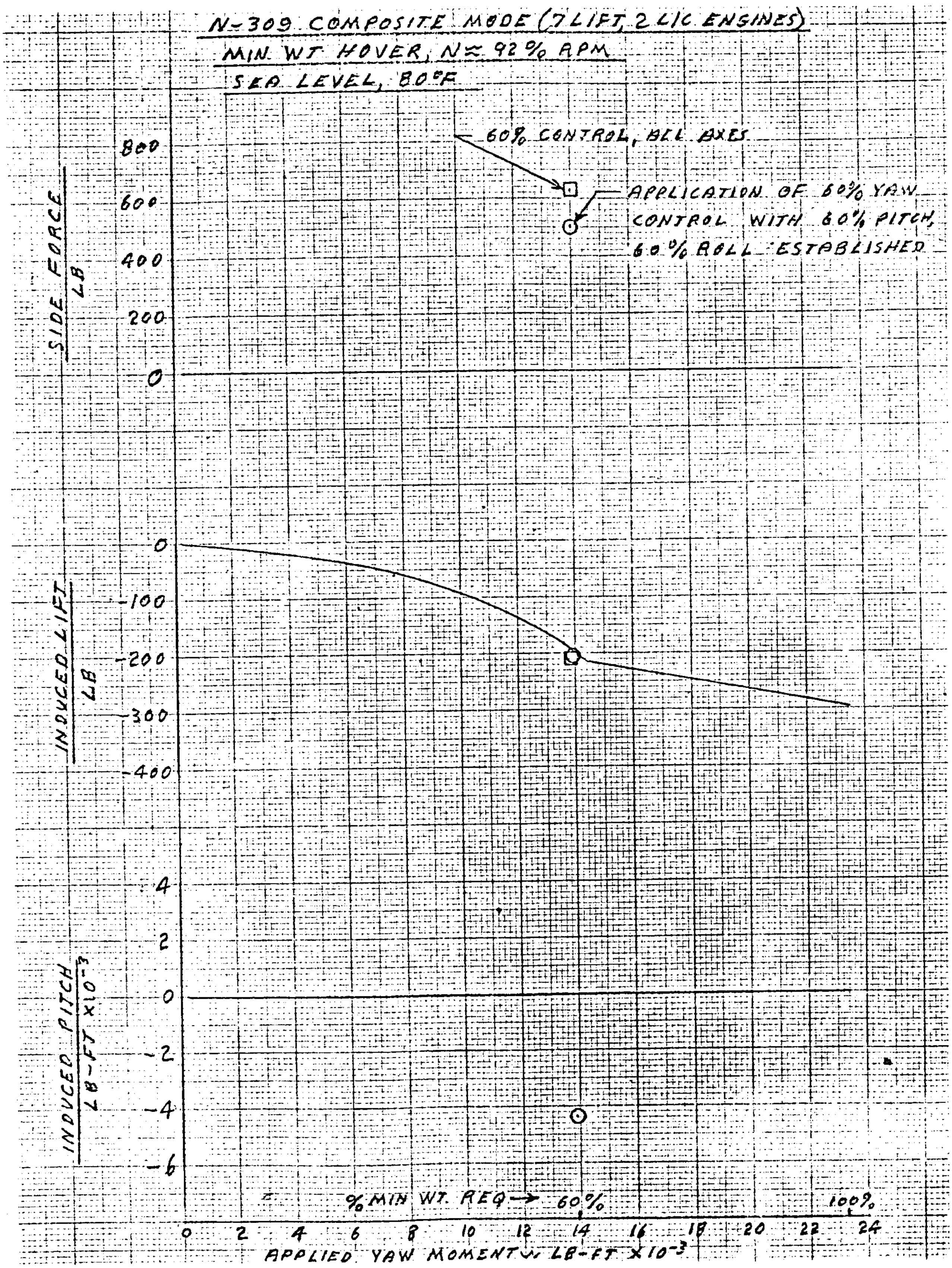


FIGURE 4-21. PITCH AND LIFT CROSS-COUPLING AND SIDE FORCE RESULTING FROM YAW CONTROL DURING HOVER

additional pitching moment of 4000 lb-ft, which requires compensation by the pilot to retain the same pitching moment. This is well within the capability of the pitch control system since, as previously shown, the available pitch control for the stabilized 60-60-60 control case was more than adequate which includes this induced pitching moment.

The above data was directly applicable to only the composite-lift airplane. However, the only major difference between the direct-lift control system and the composite lift is in the roll system. In the direct-lift system, the induced lift loss will be proportional to the applied roll moment, attaining about 300 lb for full roll application. There is no induced pitch moment due to applied roll moment.

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5.0 STRUCTURES

5.1 STRUCTURE - MODEL N-309

The N-309 wing, fuselage, and empennage preliminary structural arrangements are shown in Figures 5-1 (AD 4501), 5-2 (AD 4500), 5-3 (AD 4502), respectively. The cockpit enclosure is shown in Figure 5-4 (AD 4504). A description delineating the details of each major component follows.

5.1.1 Wing Group

The wing is a full cantilever all metal structure, consisting of a basis wing box structure, leading edge and trailing edge flaps, ailerons, trunnion supports for the main landing gear and all necessary ducting, fairing, and supports for mounting the lateral reaction jet controls at the wing tips. Access doors are provided for servicing all controls and movable surface activators.

5.1.1.1 WING. The basic structure of the main wing box consists of a two-spar multi-rib type construction with upper and lower stringer stiffened covers. The front spar is located at the 20 percent line and the aft spar at the 54 percent line. The ribs are spaced at 10 to 11 inch intervals. An auxiliary spar is located at the 70 percent line and is used to mount the supports for the trailing edge flaps and the ailerons. The center section wing box consists of front and rear spars and upper and lower stringer stiffened covers. The center section and outboard panel wing box spars and skin panels are structurally joined at the wing trunnion attaching ribs. An external doubler is used to reinforce the skin joint at the trunnion rib. The wing attaches to the fuselage by four bolts connecting the wing trunnion ribs to four trunnion fittings mounted on the fuselage lower longerons.

The main landing gear trunnion is supported at the landing gear trunnion support rib and the rear spar of the wing box. The rear spar is structurally joined to the trunnion support rib to form an integral support for the landing gear trunnion.

The roll control air ducting, with internal duct temperature of approximately 500 F, enters the wing box at the ℄ of the airplane, branching into the center of each wing box and terminating at the roll control nozzle located in each wing tip

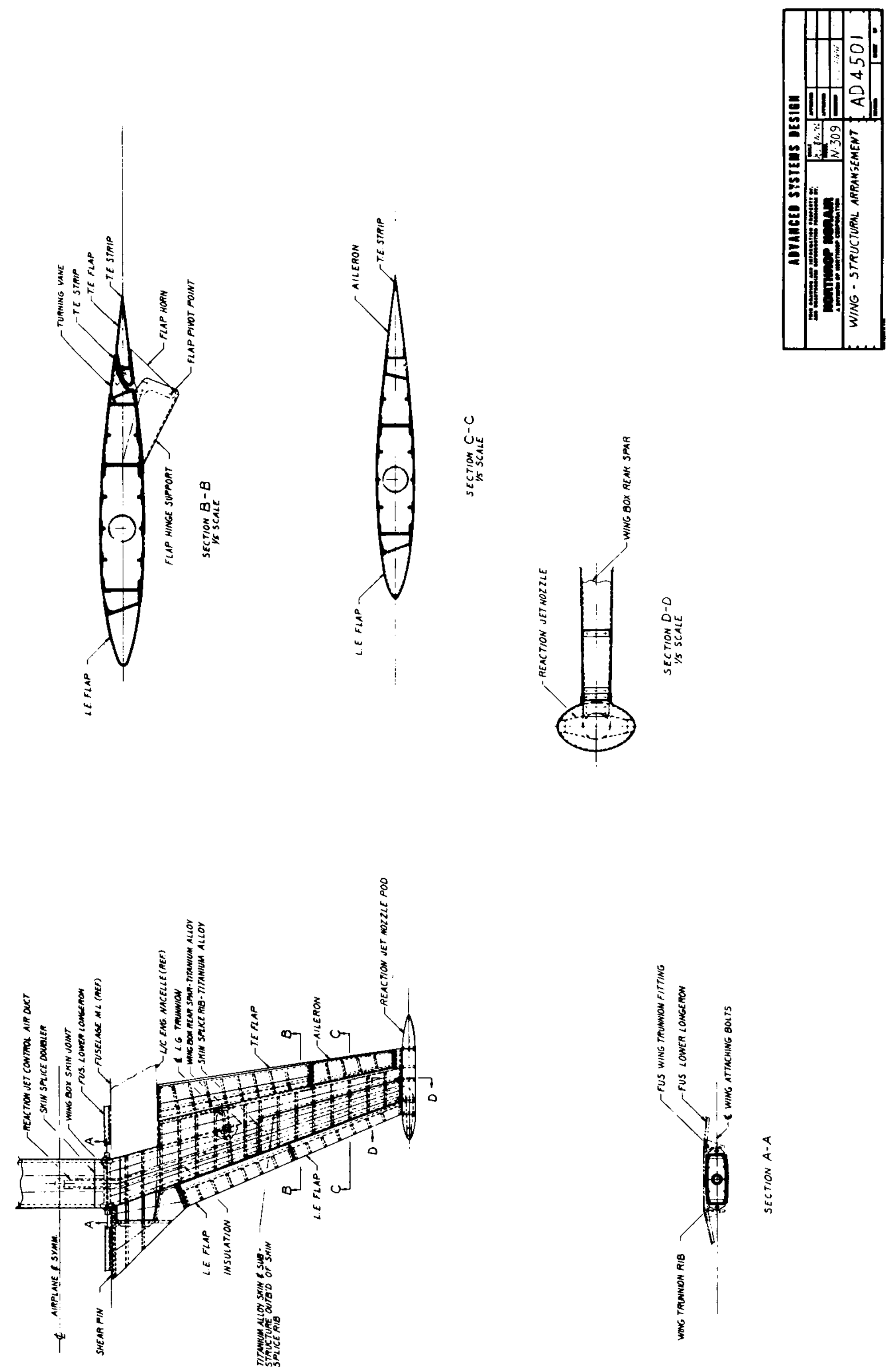
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
N-309
WING - STRUCTURAL ARRANGEMENT
AD4501
TURNING VANE
TE STRIP
TE FLAP
FLAP HORN
FLAP PIVOT POINT
FLAP HINGE SUPPORT
LE FLAP
SECTION B-B 1/5 SCALE
AILERON
SECTION C-C 1/5 SCALE
WING BOX REAR SPAR
REACTION JET NOZZLE
SECTION D-D 1/5 SCALE
REACTION JET CONTROL AIR DUCT
SKIN SPLICE DOUBLER
WING BOX SKIN JOINT
FUS. LOWER LONGERON
FUSELAGE M.L (REF)
L/C ENG NACELLE (REF)
℄ L.G. TRUNNION
WING BOX REAR SPAR-TITANIUM ALLOY
SKIN SPLICE RIB-TITANIUM ALLOY
REACTION JET NOZZLE POD
AIRPLANE ℄ SYMM.
SHEAR PIN
INSULATION
TITANIUM ALLOY SKIN & SUB-STRUCTURE OUTB'D OF SKIN SPLICE RIB
FUS WING TRUNNION FITTING
FUS LOWER LONGERON
℄ WING ATTACHING BOLTS
WING TRUNNION RIB
SECTION A-A

FIGURE 5-1

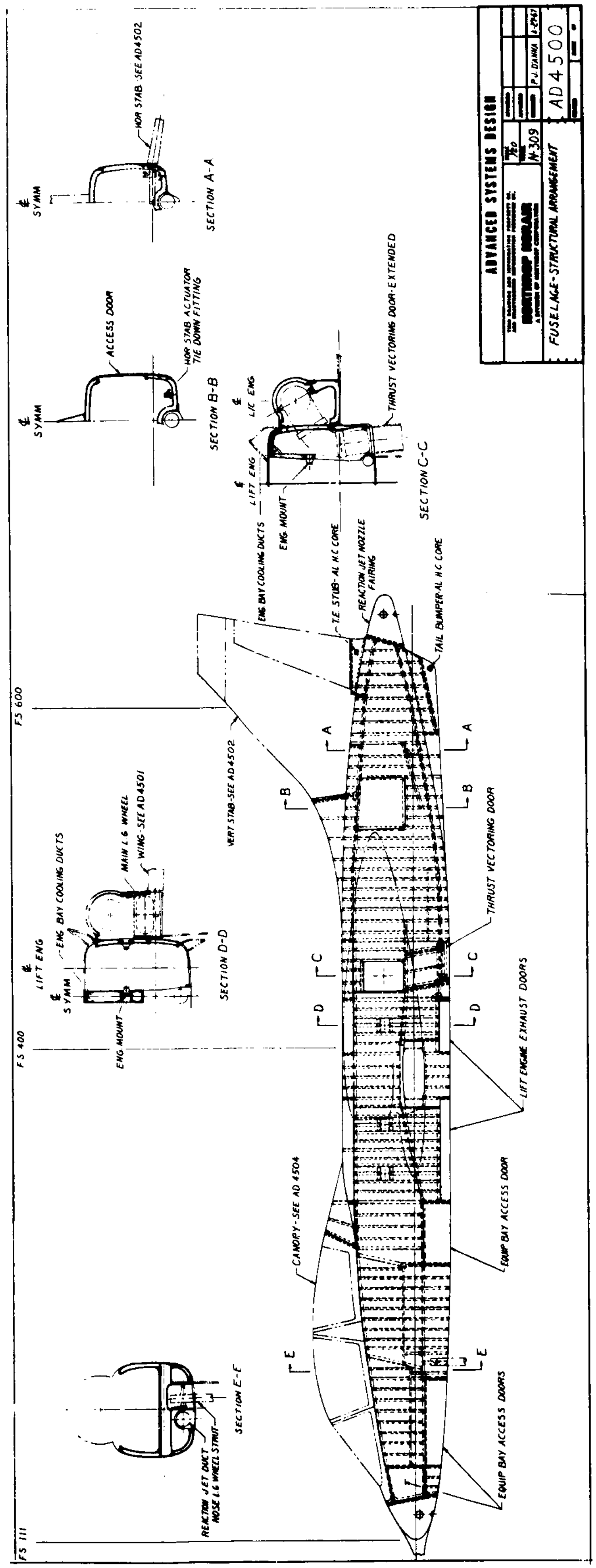
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
N-309
AD4500
FUSELAGE-STRUCTURAL ARRANGEMENT
SECTION A-A
SECTION B-B
SECTION C-C
SECTION D-D
SECTION E-E
ACCESS DOOR
HOR STAB ACTUATOR TIE DOWN FITTING
THRUST VECTORING DOOR-EXTENDED
ENG BAY COOLING DUCTS
ENG MOUNT
TAIL BUMPER-AL HC CORE
VERT STAB-SEE AD4502
THRUST VECTORING DOOR
LIFT ENGINE EXHAUST DOORS
EQUIP BAY ACCESS DOOR
CANOPY-SEE AD4504
EQUIP BAY ACCESS DOORS
FS 111
FS 400
FS 600

FIGURE 5-2

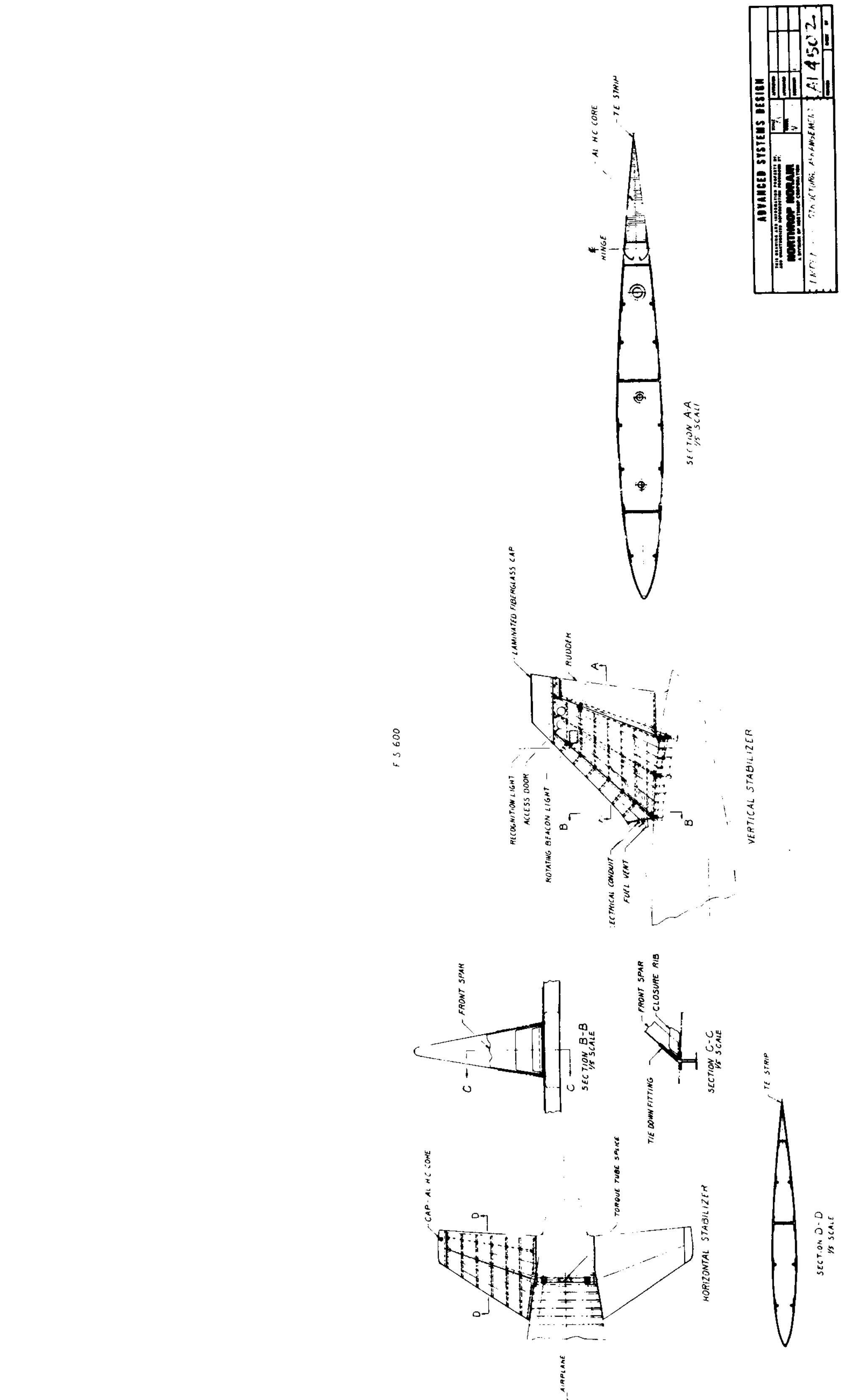
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
SECTION A-A
1/5 SCALE
HINGE
AL HC CORE
TE STRIP
LAMINATED FIBERGLASS CAP
RUDDER
F.S. 600
RECOGNITION LIGHT
ACCESS DOOR
ROTATING BEACON LIGHT
FUEL VENT
VERTICAL STABILIZER
FRONT SPAR
SECTION B-B
1/5 SCALE
FRONT SPAR
CLOSURE RIB
SECTION C-C
1/5 SCALE
TIE DOWN FITTING
TE STRIP
CAP - AL HC CORE
TORQUE TUBE SPLICE
HORIZONTAL STABILIZER
SECTION D-D
1/5 SCALE
AIRPLANE

FIGURE 5-3

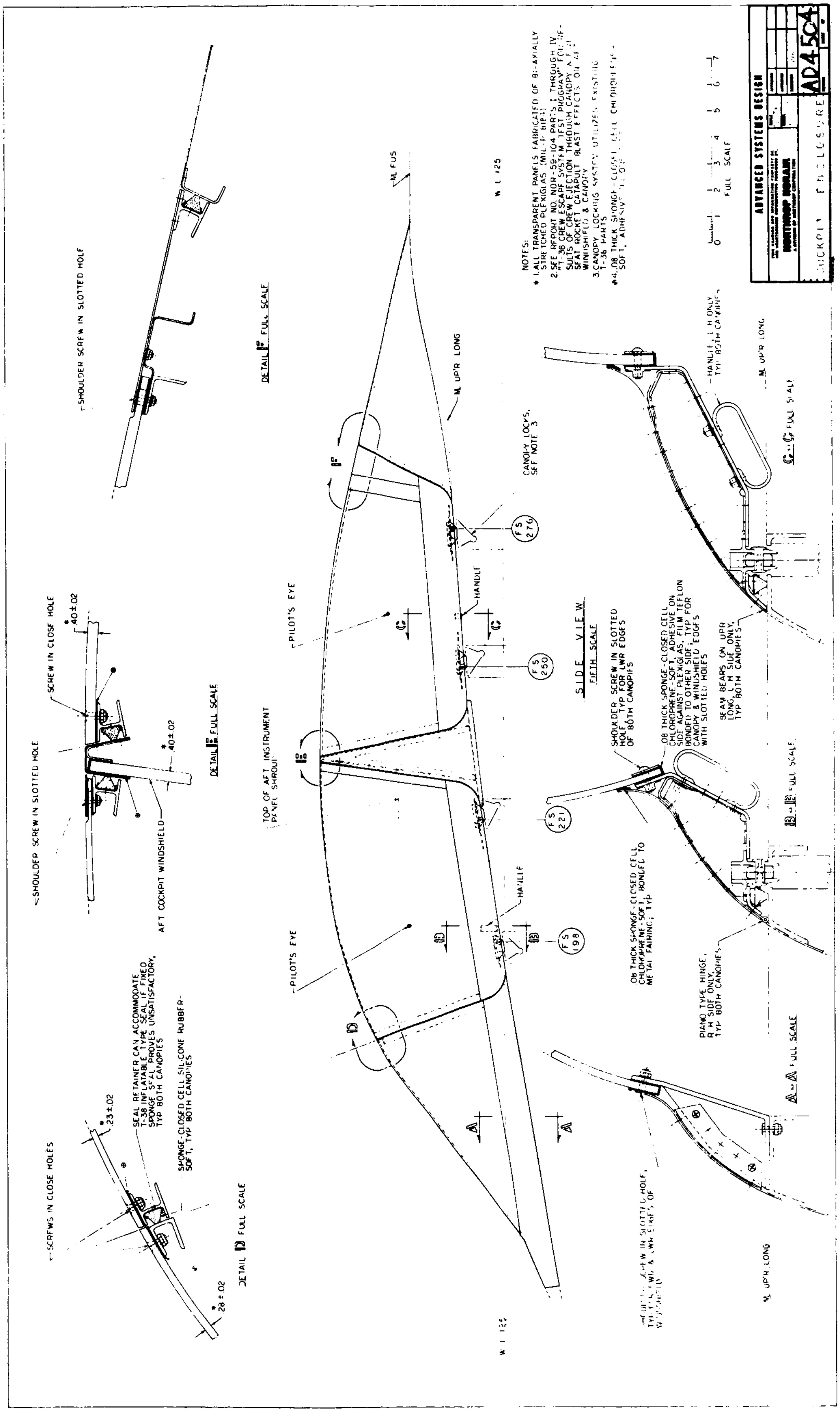

FIGURE 5-4

fairing. Insulation is wrapped around the air ducting to a point, a few inches outboard of W.S. 125 where, due to limited clearance with the substructure, it is terminated. In this area of the wing box, outboard of W.S. 125, where structural temperatures are expected to exceed 220 F, titanium alloy skins and substructure are used. The forward half of the roll control nozzle fairing is made of stainless steel while the aft half is made of aluminum alloy. All skin panels and substructures where temperatures are not expected to exceed 220 F are made of 7075-T6 aluminum alloy.

An extended leading edge fairing forms the inboard leading edge of the wing, terminating inboard of the leading edge flap. A shear pin forms the only structural tie between the inboard side of the fairing and the fuselage.

5.1.1.2 AILERONS. The ailerons are of conventional design and construction utilizing 7075-T6 aluminum alloy material. The basic structure consists of a front spar, upper and lower skins, a trailing edge strip and ribs spaced at approximately 10 inch intervals. The aileron is attached to the wing by means of a piano type hinge mounted to the upper surface of the rear spar of the wing. The ailerons will be balanced statically and dynamically as determined by analysis.

5.1.1.3 TRAILING EDGE FLAPS. The trailing edge flaps are of conventional design and construction utilizing 7075-T6 aluminum alloy material. The basic structure consists of a front spar, leading edge and trailing edge ribs, spaced at approximately 10 inch intervals, contoured leading edge skin, and upper and lower aft skins, splicing at the front spar and a trailing edge strip. The flaps are externally hinged, to provide the required motion, at three locations along its span. Anti-friction type ball bearings are used at each pivot point. The flaps are supported by means of three flap hinge supports externally mounted to the aft portion of the main wing structure. The trailing edge flaps extend from the outboard side of the L/C engine nacelle to the inboard side of the ailerons.

5.1.1.4 TURNING VANES. The basic structure consists of a formed front spar, ribs spaced at approximately 10 inch intervals, upper and lower skins and a trailing edge strip. All materials are 7075-T6 aluminum alloy. The vanes rotate about the ℄ of a piano type hinge mounted at the forward lower portion of the vane front spar and the lower flange of the rear spar of the main wing structure. The motion of the vane is slaved to that of the flap by three pushrods having anti-friction ball bearing rod ends. The upper end of each push rod is attached to a rib of the turning vane by means of a fitting while the lower end attaches to a flap hinge fitting. The turning vanes extend from the outboard side of the L/C engine nacelle to the inboard side of the ailerons.

5.1.1.5 LEADING EDGE FLAPS. The basic structure of the leading edge flaps consists of a spar, ribs spaced at approximately 10 inch intervals and a contoured leading edge skin. All materials are 7075-T6. The flap pivots about the ℄ of a piano type hinge mounted on the lower flanges of the flap spar and the forward spar of the wing box. The leading edge flaps are made in two sections with the inboard section extending to approximately W.S. 80 and the outboard section extending to a point just inboard of the reaction control nozzle fairing.

5.1.2 Body Group

The body group consists of the fuselage structure and all integral provisions for attachment of the wing, empennage, left/cruise engine nacelles, installation of the lift engines, crew-compartment, subsystems, fuel and equipment.

5.1.2.1 FUSELAGE. The arrangement of the fuselage consists of a nose section housing the flight test boom and pitch/yaw control nozzle; a forward section housing the crew compartment, nose landing gear and equipment and/or payload; a center section housing the lift engines, fuel cells, wing carry through structure and attaching structure for the lift/cruise engine nacelles and an aft section supporting the empennage, tail bumper and pitch/yaw control nozzle fairing.

The basic structure consists of upper and lower longerons, skin panels and vertical frames or bulkheads spaced at approximately 6 inch intervals to minimize acoustically induced fatigue failure of the skins. In the forward section of the fuselage three equipment bays are provided, one aft of the pitch/yaw control nozzle fairing, one below the front cockpit floor and another aft of the rear cockpit bulkhead. A fuel bay floor is provided at the level of the lower longeron in the area of the aft equipment bay. Access to the fuel bay is through the equipment bay by means of stressed doors provided in the fuel bay floor. The nose wheel trunnion mounts on two fittings attached to two longitudinal beams which also form the structural housing into which the gear retracts. The wheel door is mounted to a piano type hinge which attaches to the outboard trunnion support beam. The ℄ of the nose wheel strut is offset six inches to the left of the airplane ℄. Glass fiber matting is provided between the inner and outer skin panels of the crew compartment to insulate the cockpit area from the noise environment. In the center section of the fuselage, intermediate bulkheads divide the engine bay into four compartments for housing the lift engines. Two lift engines are housed in each of the forward three compartments and one lift engine plus the lift exhaust nozzles from the L/C engines in the aft compartment. Supporting structure is mounted between bulkheads for attaching the lift engine support mounts and a vertical longitudinal firewall between each pair of engines. Double hinged air inlet doors are provided over each lift engine. These doors are attached to the upper longerons by means of a piano type hinge. Double hinged exhaust doors are provided for the lift engines in the forward three compartments. These doors are attached to the lower longerons by means of a piano hinge. Bomb-bay type exhaust doors are provided for the aft lift engine. These doors are mounted on structural members, with integral piano-type hinges, attaching to the forward and aft bulkhead of the engine compartment. Thrust vectoring doors are provided for the lift exhaust nozzles of the L/C engines. Hinge supports, containing anti-friction bearings, for these doors mount on an auxiliary bulkhead. A reinforced opening is provided at each side of the fuselage to permit entry of the lift exhaust nozzles of the L/C engines and supporting structure provided for mounting the nozzle to the fuselage. Supporting structure is also provided for attaching the L/C engine nacelles to the fuselage. The wing support trunnion fittings are mounted to the lower longerons. Removable access panels are provided to permit access to engine controls, accessories and plumbing and to the fuel bay over the wing box.

The aft section of the fuselage consists of supporting structure for mounting the pitch/yaw control nozzle fairing, the tail bumper and the horizontal and vertical stabilizers. The lower portion of the tail bumper incorporates a removable aluminum honeycomb core stub which may be easily replaced if damaged during a tail first landing. The torque tube of the horizontal stabilizer is supported by two anti-friction type bearings mounted in housings attached to each side of the fuselage. The three tie down fittings of the vertical stabilizer pass through the top skin of the fuselage and attach to the upper webs and flanges of their supporting fuselage frames. Access doors are provided on the top of the fuselage, at each side of the rudder post to permit servicing of the rudder actuators. A removable panel is provided aft of the lift/cruise engine nacelle to permit entry to the fuel and equipment bays and to permit servicing of the horizontal stabilizer actuators. Adequate removable panels are provided on the underside of the fuselage, in line with the control air ducting to permit easy inspection and servicing.

5.1.2.2 LIFT/CRUISE ENGINE NACELLES. The arrangement of each engine nacelle incorporates a nose section housing the air inlet duct, a center section housing the engine and its accessories and an aft fairing housing the exhaust tail pipe.

The basic structure consists of upper and lower longerons, skin panels with frames and intercostals spaced so as to minimize acoustically induced fatigue failure of the skins, and supporting structure for attaching the nacelle to the fuselage. The upper portion of the nacelle, in the vicinity of the engine, is made removable to permit engine installation and/or removal. The lower main engine mount is supported on the upper outboard longeron while provisions are made for attaching the upper mount to the sub-structure of the engine removable access panel. Firewalls are provided around the periphery of the engine and removable access panels added to permit access to critical engine controls and accessories. The forward portion of the nacelle passing over the wing box is not structurally attached to the wing in order to prevent any wing loads from being transfered to the nacelle structure.

The major portions of the skins and substructure of the nacelles and fuselage are made of 7075-T6 aluminum alloy. Firewalls, pitch/yaw control ducting and nozzle fairings, engine compartment bulkheads are made from stainless steel alloy material. The thrust vectoring doors and local structure where temperatures may exceed 220 F are made of other heat resistant material.

5.1.3 Empennage

The empennage group consists of a fixed vertical stabilizer mounting a rudder assembly and a movable horizontal stabilizer, fuselage mounted. It also includes all provisions for mounting the surfaces to the aft section of the fuselage.

5.1.3.1 HORIZONTAL STABILIZER. The horizontal stabilizer consists of a left and right hand assembly integrally connected by torque tubes which are structurally joined at the ℄ of the airplane. The torque tube is supported by anti-friction type ball bearings mounted in bearing housings attached to each side of the fuselage. An eight inch radius horn fitting mounts on the torque tube of each stabilizer assembly and these are attached to the rod end of the hydraulic actuators used for moving the surface.

The basic structure of the stabilizer consists of a 50 percent line spar multi-rib type of construction with upper and lower stringer stiffened covers. The leading and trailing edge covers are spliced at the spar caps. In addition the trailing edge covers are joined to an aluminum trailing edge strip which runs the span of the trailing edge up to the tip caps. The tip caps are of aluminum honeycomb core construction while all skins and substructure of the stabilizer are of 7075-T6 aluminum alloy material.

5.1.3.2 VERTICAL STABILIZER. The vertical stabilizer consists of a fixed cantilevered fin mounting a rudder assembly. A removable laminated fiberglass tip cap is provided for housing the VHF antennas. A recognition light and rotating beacon light is mounted below the tip cap. A fuel vent tube is installed for venting the fuel tanks past the trailing edge of the fin at a point above the rudder.

The basic structure of the fin incorporates a leading edge assembly and a main box assembly. The leading edge assembly consists of a contoured stringer stiffened cover and ribs spaced at approximately 12-inch intervals. The basic structure of the main box structure consists of a three spar multi-rib type of construction with left and right hand stiffener reinforced covers. The front spar attaches to the leading edge skin and ribs while the rear spar incorporates a hinge fitting for mounting the rudder. The other rudder support hinge fitting is attached to fuselage frames at the ℄ of the airplane. All rudder hinges incorporate anti-friction type or ball bearings. The three spars attach to the fuselage frames by means of tie down fittings while a closure rib completes the structural tie between fin and fuselage.

5.1.3.3 RUDDER. The basic rudder consists of a front spar, trailing edge strip, two end ribs and skin covers all bonded to an aluminum honeycomb core filler. The rudder post, an integral part of the front spar, incorporates a horn fitting with two diametrically opposite attach points for the rod ends of the hydraulic actuators used for moving the surface.

The skins and substructure of fin and rudder, except for the tip cap and honeycomb core are made of 7075-T6 aluminum alloy.

5.1.4 Cockpit Enclosure

The cockpit enclosure consists of a contoured one-piece windshield for the forward cockpit and two one-piece canopies, one for each crewman. In addition, a one-piece flat windshield is provided for the aft cockpit to protect the aft crewman from the wind and rocket blast exhaust resulting from ejection of the forward seat. Each canopy is hinged along the right hand edge of the cockpit opening to provide easy access to the cockpit. Locks and handles are provided on the left side of each canopy. Both canopies may be locked or opened from both the inside and outside of the cockpit enclosure. The cockpit locking system utilizes existing T-38 parts. The area in line with the ejection path of the crew seats contains no metal framework in order to permit safe ejection through the canopy. All transparent panels are fabricated of bi-axially stretched plexiglass (MIL-P-8184).

5.2 STRUCTURE - T-39A MODIFICATION

The T-39A structural diagram of the modified and unmodified fuselage and wing are shown in Figure 5-5 (AD 4491). A description delineating the major modifications to the wing, fuselage and empennage are given below.

5.2.1 Wing

A hole is provided through the center section wing box and substructure to accommodate two of the lift engines. External doublers and local reinforcements are added to retain structural integrity. The inboard end of the trailing edge flaps are shortened approximately 12 inches to clear the new engine nacelles. The wing ribs are modified to permit installation of the reaction control ducting. Other modifications include: removal of existing wing tip cap and modifications of local structure to permit installation of control nozzle fairing; installation of trunnion supports and strut doors for new landing gear.

5.2.2 Forward Fuselage

Seventeen inches were added to the length of the nose to accommodate pitch/yaw control nozzle. Other modifications include: a flight test boom installed; new canopy and windshield; new ejection type seats, instrument panel and consoles, a new nose gear and wheel fairing.

5.2.3 Center Fuselage

A 24-inch section was added to the fuselage forward of F.S. 206. Other modifications include: removal of baggage and cabin compartment floors, emergency escape hatch, entry doors and cabin windows; fuselage structure modifications between F.S. 143 and F.S. 333.6, to accommodate lift engines, by adding new frames, firewall bulkheads, provisions for engine mounts, and intake and exhaust doors; upper longeron removed in left engine bay area and replaced by new relocated longeron. In addition to the above, the existing cruise engine nacelles are removed and replaced by new nacelles located forward of the existing ones.

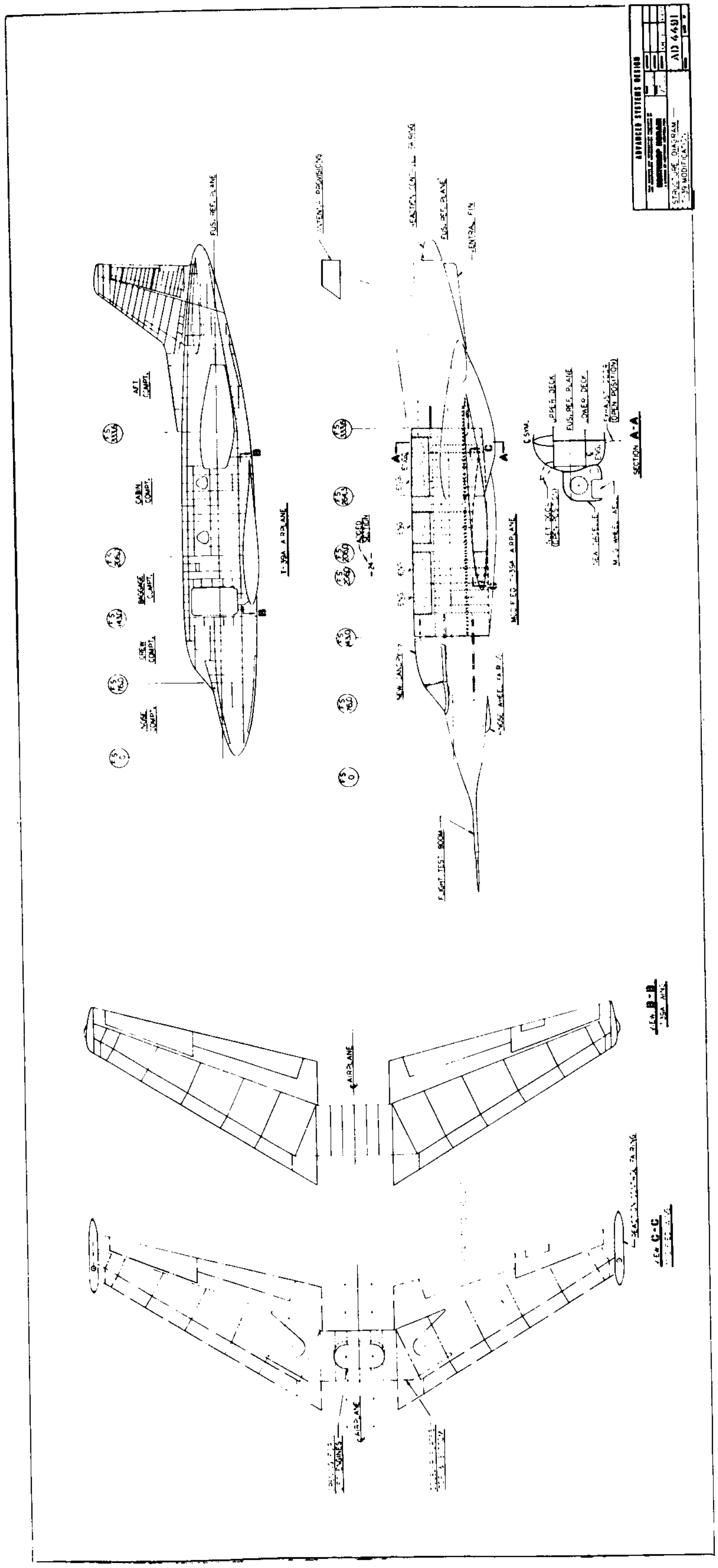

FIGURE 5-5

5.2.4 Aft Fuselage

Modifications to the fuselage aft of F.S. 333.6 include: structural modifications for additional horizontal stabilizer deflection requirements and support provisions for new longer stroke actuator; longeron, frame and stiffener modifications due to removal of existing cruise engine nacelles and replacement by new nacelles; structural provisions for installing tail bumper and reaction jet control nozzle fairing.

5.2.5 Vertical Stabilizer

The existing fin tip cap is removed and local structure modified to accommodate new laminated fiberglass cap housing VHF antennas.

5.3 MATERIALS

All new structural components where temperatures are not expected to exceed 220 F will be made of 7075-T6 aluminum alloy material. Firewalls and control air ducting and nozzle fairings will be constructed from stainless steel alloy material. All other local structure where temperatures are expected to exceed 220 F will be constructed of other heat resistant alloy material.

Special attention is given to materials in areas subjected to an elevated temperature and acoustic environment. The final selection of these materials will include considerations of cost, fabrication and reliability. The high temperature environment varies from 500 F hot gas containment to 1320 F exhaust gas impingement for durations exceeding five minutes. The primary areas of high temperatures are engine bay compartment, reaction control ducts and nozzles including local structure in which they are housed, and thrust deflection doors. While use of titanium will be generally avoided because of its high cost, titanium alloys will be considered in critical weight areas. The materials considered for design are discussed below.

5.3.1 Aluminum Alloys - Up to 220 F

The major portion of the primary structure, where structural temperatures are not expected to exceed 220 F will be fabricated from 7075-T6 aluminum alloy material. Within this temperature range this alloy possesses excellent strength to weight ratios. In addition, the yield stresses are only approximately 13 percent below the ultimate stress. Therefore a vehicle stressed to 100 percent of limit design load (where the allowable limit design stresses are equal to the ultimate stress of the alloy divided by 1.5), will posses a margin of safety over the allowable yield stress of better than 30 percent.

5.3.2 Titanium Alloys - Up to 600 F

The alpha-beta alloys, e.g., Ti-6Al-4V and Ti-8Mn have very good strength at room temperature and posses good elevated temperature strength and stability up to about 600 F.

5.3.3 Precipitation Hardening Stainless Steels - Up to 1200 F

Precipitation hardening stainless steels are excellent for high temperature in an oxidizing atmosphere. The basic alloys considered are:

(a) PH 14-8Mo: a higher toughness and fatigue strength version of the better known PH 15-7 Mo. This alloy is weldable and formable.

(b) AFC-77: a maraging stainless steel with excellent strength and oxidation resistance up to 1200 F.

(c) Austenitic Stainless Steels or "18-8" Stainless: an excellent low cost weldable, formable material with good errosion resistant properties.

5.3.4 Superalloys - 1200 F to 1800 F

Exposure above 1200 F for extended periods of time with high loads requires alloys with a nickel or cobalt base. Superalloys are planned for the thrust deflector doors. The standard alloys considered in these areas are Rene' 41 and Inconel X. Both are weldable and formable within certain limits.

5.3.5 Thermal Insulation

The containment of the thermal environment can be accomplished by utilizing metals with high reflectivity and by using materials of low density and low thermal conductivity. The latter materials include fibrous mats of silica, alumina, and asbestos fibers for temperatures to 1300 F. Low thermal conductivity insulation is wrapped around the control air ducting and utilized between interface areas of steel and aluminum.

5.4 STRUCTURAL DESIGN CRITERIA

5.4.1 Maneuver Load Factors

New aircraft will be designed to limit maneuver load factors of + 3.75 g and - 1.5 g. In flight operations, the aircraft will be restricted to 80 percent of these load factors.

Design load factors are not specified for modified aircraft. The most feasible acceptable limit load factors will be determined for the modified aircraft. Existing structure of modified aircraft, which has been static tested, can be operated to 100 percent of the allowable limit load factors determined. New structure added to the modified aircraft will be designed to 1.25 times the allowable load factors.

5.4.2 Airspeed

Maneuver loads for new aircraft will be based on a maximum speed envelope of 450 KEAS (q = 685 PSF) or 0.80 Mach, whichever is lower. The 450 KEAS defines the speed envelope from sea level to 9,000 feet altitude. Above 9,000 feet the speed envelope is defined by the 0.80 Mach number (refer to Figures 5-6 and 5-7).

The most feasible speed envelope will be determined for modified aircraft to be used in conjunction with allowable maneuver load factors.

5.4.3 Atmospheric Gust

To avoid excessive loads resulting from atmospheric turbulence, the following analyses will be conducted:

1. Speed restrictions will be determined so that the loads resulting from a 50 fps vertical gust (U_{d_e}) do not exceed the allowable operating loads.
2. Using the maneuver speed envelopes, the allowable gust velocities will be determined based on the allowable operating loads.
3. The speed restrictions imposed by Item 1 must result in an adequate operating envelope to satisfy the primary objectives of the research vehicles.

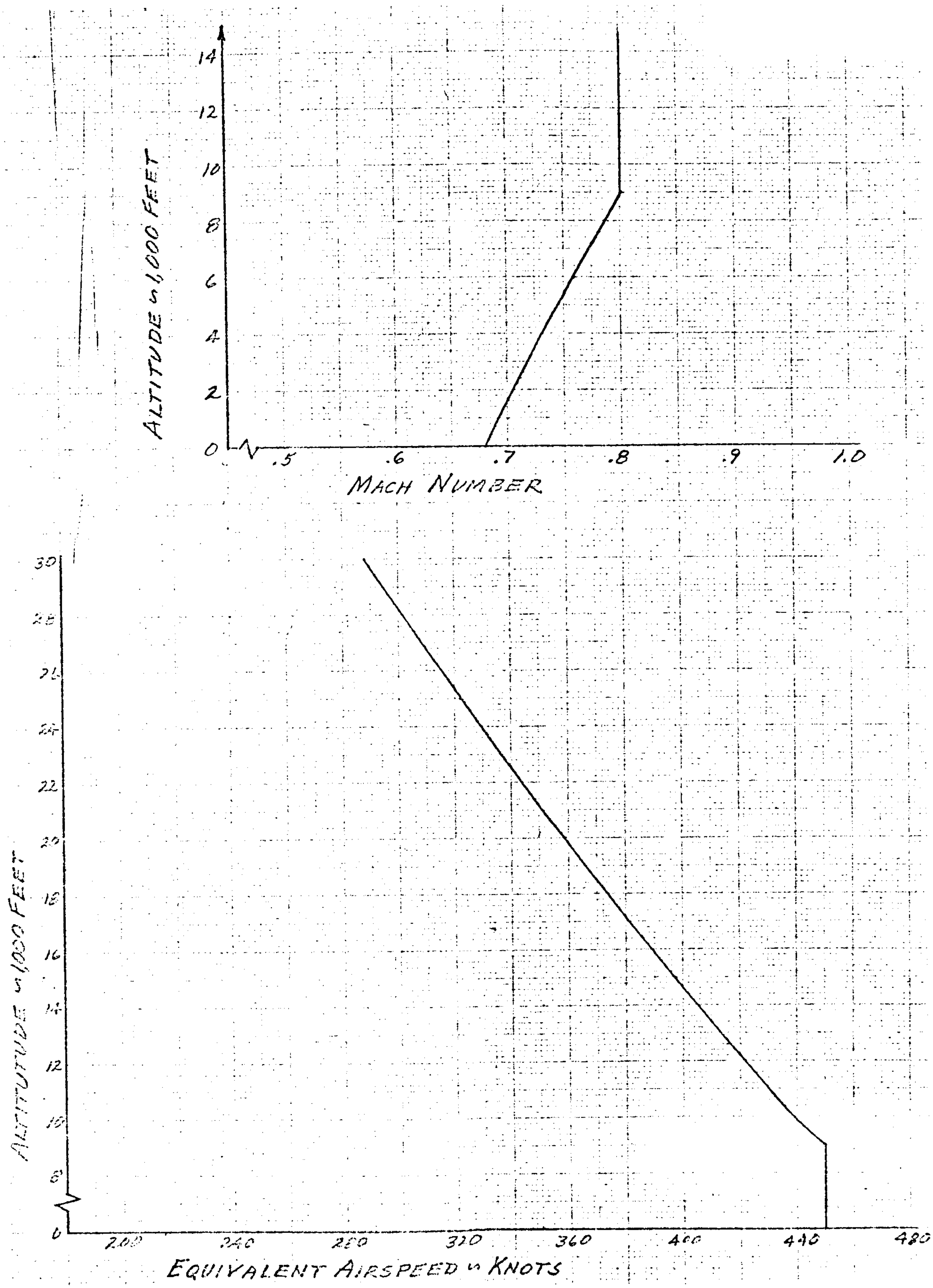

FIGURE 5-6. MAXIMUM MACH AND EQUIVALENT AIRSPEED VS. ALTITUDE

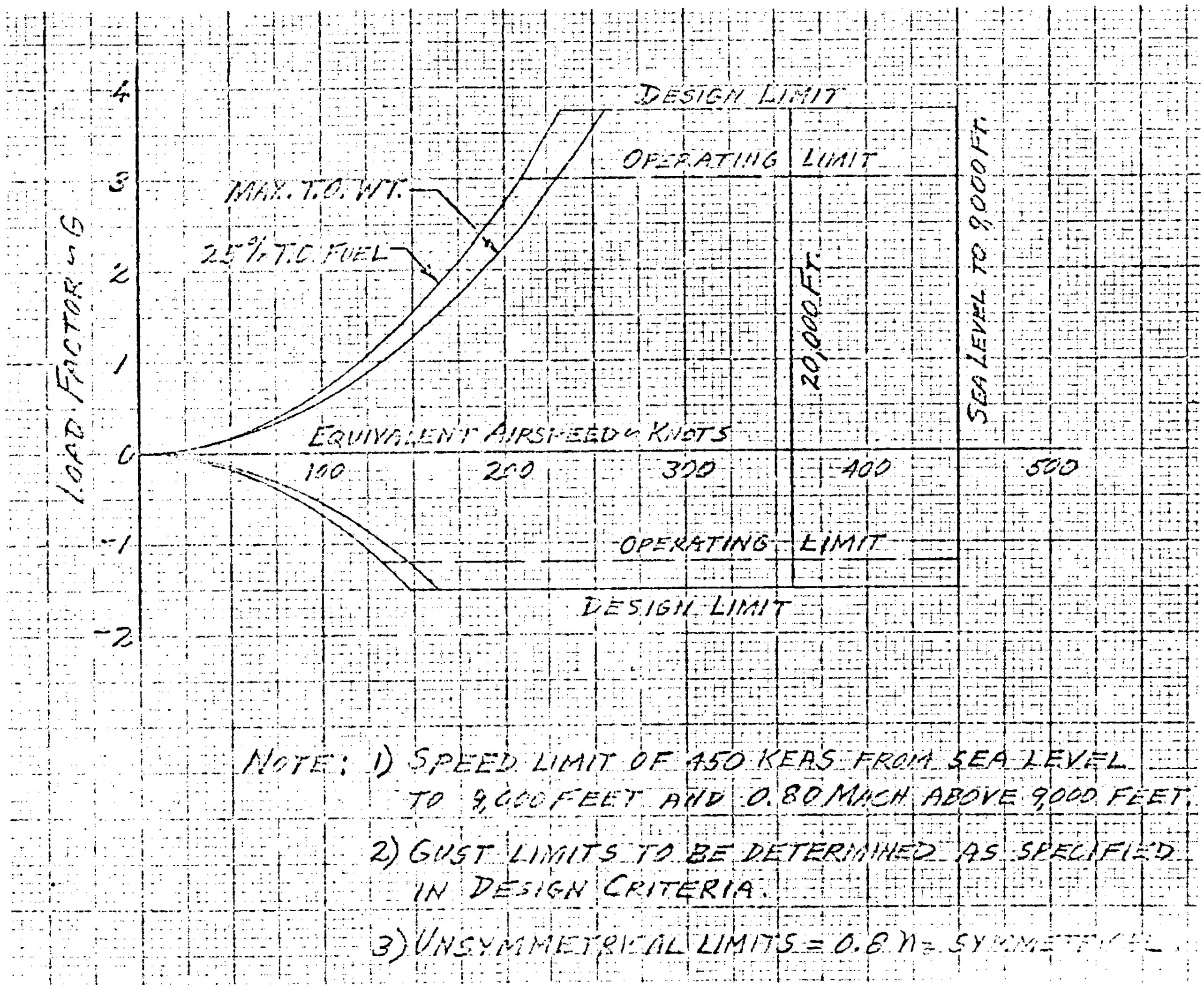

FIGURE 5-7. DESIGN AND OPERATING LIMITS FOR SYMMETRICAL MANEUVER

5.4.4 Landing Loads

The structure will be designed to the loads resulting from conventional and vertical landings. The method of analysis and criteria of MIL-A-8862(ASG) will be used where applicable.

The following landing conditions will be used for design of new aircraft:

1. Conventional landing - 12 fps sink speed with 1.0 g lift.

2. VTOL landing - 15 fps sink speed with 2/3 hovering thrust. The following conditions will be considered:

   a. The aircraft structure (except for the landing gear) will be designed for a 15 fps sink speed with a side velocity of 5 knots. The landing gear itself will be designed to the helicopter obstruction requirements of MIL-S-8698 at 15 fps vertical sink speed.

   b. Structure and landing gear will be capable of withstanding the loads produced by a VTOL sink speed of 10 fps with the aircraft in a 10 degree banked attitude.

Modified aircraft will be designed to the most feasible sink speed entailing reasonable changes.

5.4.5 Ground Handling

Loads due to towing, jacking, hoisting, and mooring will be based on MIL-A-8862(ASG).

Provisions will be made for towing by the nose wheel and will accommodate towing speeds of 20 mph maximum over smooth and hard surfaces and 3 mph maximum over soft surfaces.

Removable hold-back and tie-down fittings for engine thrust tests will be included.

5.4.6 Crash Loads

The loading conditions of MIL-A-8865(ASG) will be used to provide protection for the crew under crash conditions.

5.4.7 Pressurization Loads

Not applicable.

5.4.8 VTOL Test Stand

The aircraft will be designed to the most severe combination of loads to be encountered during ground testing on a VTOL test stand.

5.4.9 Wind Tunnel Tests

Support system attachment points for tests in the 40 by 80-foot wind tunnel at NASA Ames Research Center will be included in the design of the aircraft. A maximum dynamic pressure of 120 psf in the wind tunnel will be used. Design conditions of angle of attack, sideslip, etc. will be determined through discussions and agreement with NASA.

5.4.10 Flutter Characteristics

The aircraft will be free from divergence, flutter, buzz, or other aeroelastic instability throughout its flight and weight envelopes. MIL-A-8870 will be used as a guide for flutter requirements.

5.4.11 Service Life

Design of any new primary structure of the aircraft will be on a failsafe basis to the maximum extent feasible. Structural design will be based on a minimum service life of 300 hours with a scatter factor of 4.0 distributed in the following manner:

1. 100 hours in a conventional flight cruise at 6,000 feet altitude.
2. 100 hours at climb speed at 3,000 feet altitude.
3. 100 hours at transition speeds at 3,000 feet altitude and below.
4. 40 routine maneuvers per hour shall be included.
5. 4 V/STOL landings and transitions per flight hour shall be assumed.
6. Compatible life shall be established for components such as control surfaces, doors, etc.

5.4.12 Acoustic and Thermal Environments

The structure will be designed to the acoustic and thermal environments expected. Available test data will be used as much as possible to predict these environments.

5.5 BASIC LOADS - MODEL N-309

5.5.1 Discussion of Loads

Bending moments on the fuselage are compared on the basis of limit load for symmetrical maneuvers and conventional landing (refer to Figure 5-8). Limit bending moments are used for the comparison because landing loads are not analyzed on an ultimate basis. Ultimate fuselage shears and bending moments for symmetrical maneuver conditions are presented in Figures 5-9 and 5-10. Fuselage shears and bending moments for landing conditions are also presented in Figures 5-11 and 5-12. Landing loads are presented as design loads in accordance with paragraph 3.1.3 of MIL-A-8862(ASG). Fuselage loads for conventional landing are, generally, more severe than for vertical landing. Although the vertical landing conditions are based on a higher sink speed, they include 2/3 hovering thrust on the lift engines, which relieves the fuselage loads, and do not include the horizontal spin-up and spring-back loads which produce pitching accelerations in conventional landing. Fuselage loads for vertical landing with a side velocity of 5 knots have not been analyzed in this study phase. Loads resulting from the control jets will affect only local structure.

Wing loads are presented in Figure 5-13 for a + 3.75 g symmetrical maneuver and represent maximum bending moment on the wing. Conditions with deflected ailerons or flaps will produce wing torsion conditions but these have not been calculated at this time since the detailed aerodynamic distributions have not been determined. Forces produced by the control jets at the wing tips are not critical for the primary wing structure but must be considered for local structure.

5.5.2 Load Conditions Investigated

5.5.2.1 SYMMETRICAL FLIGHT MANEUVERS. Fuselage loads are computed for + 3.75 g and - 1.5 g assuming zero airloads on the fuselage but including a balancing

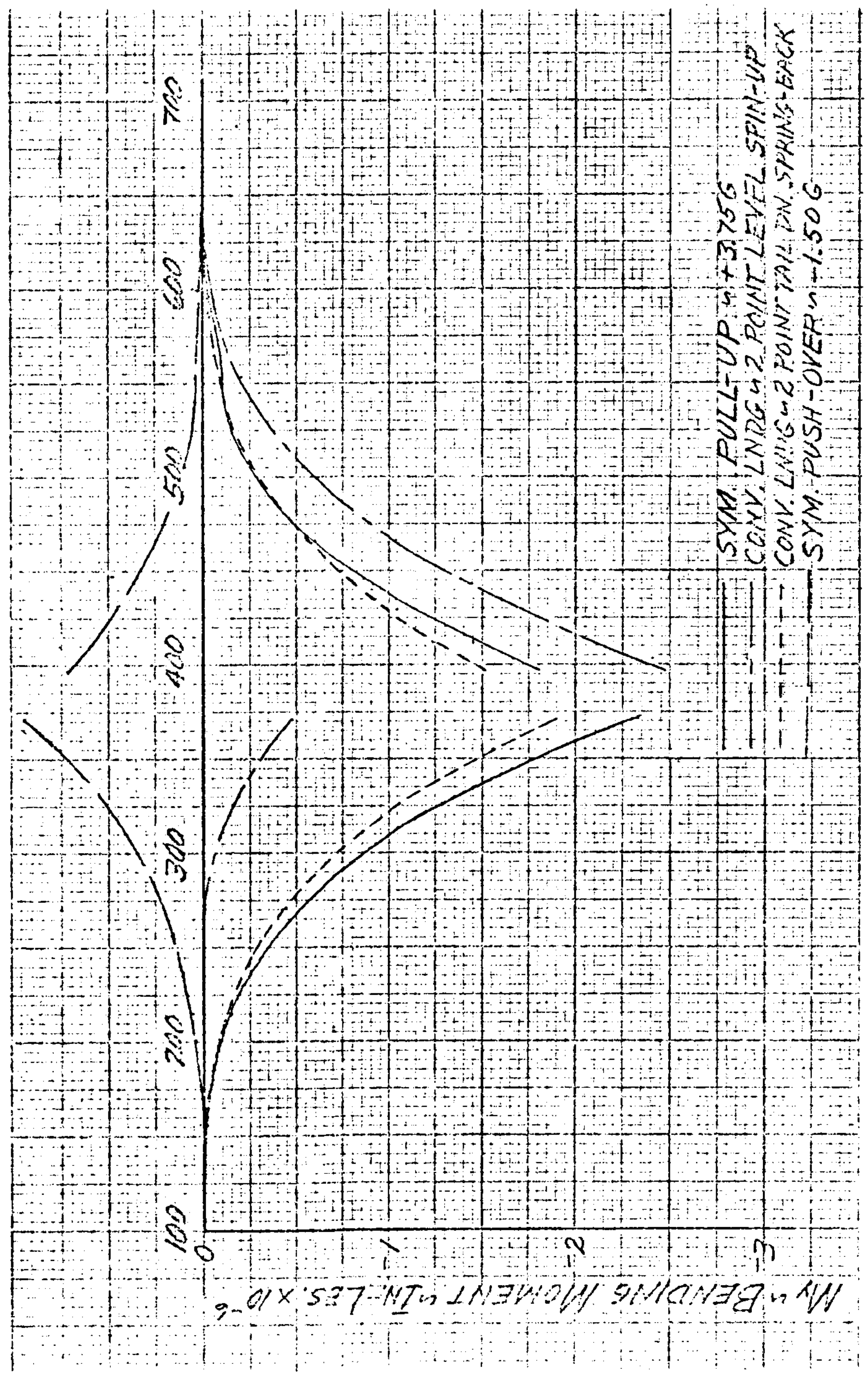

FIGURE 5-8. FUSELAGE BENDING MOMENT ENVELOPE ~ LIMIT

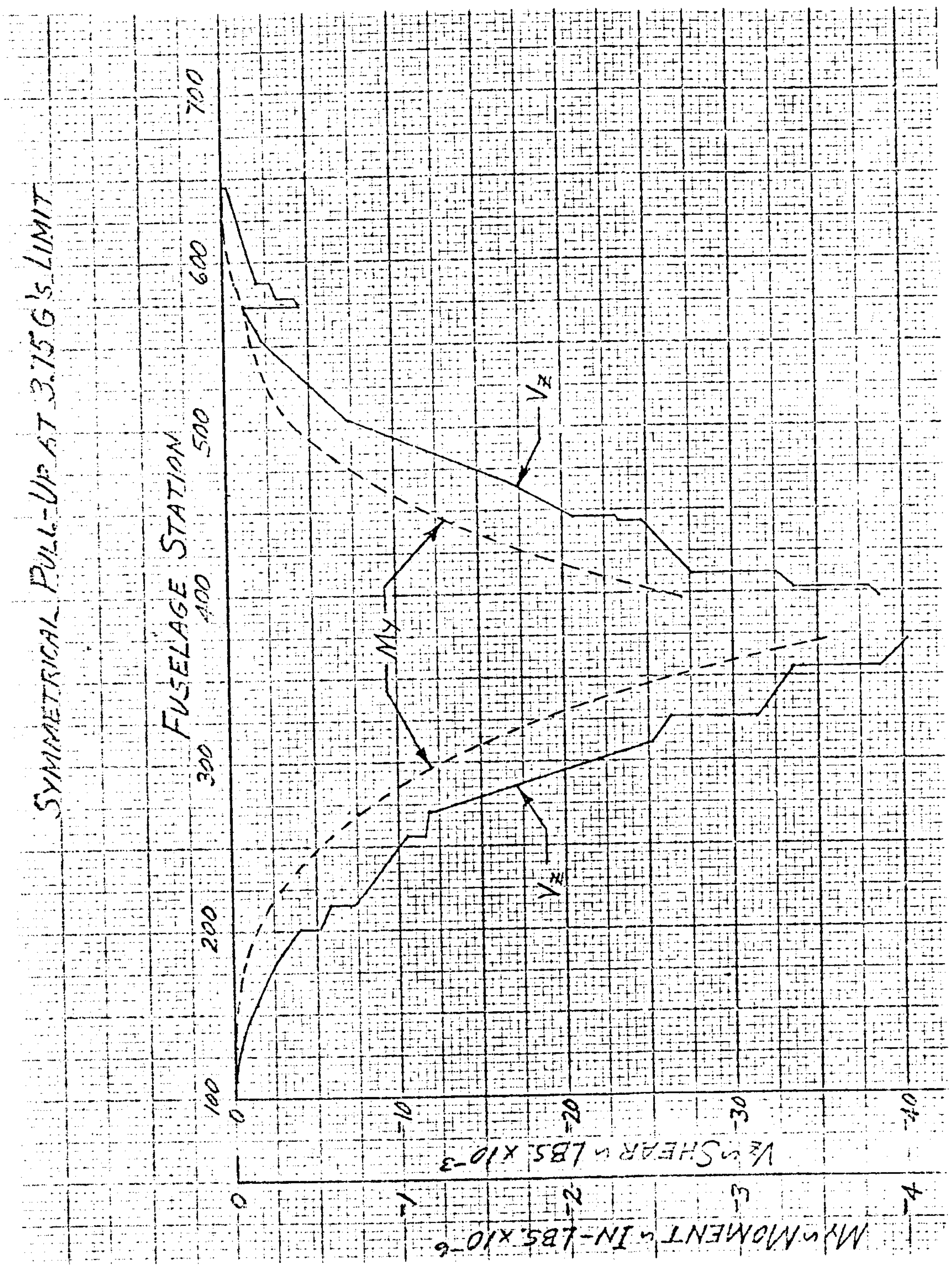

FIGURE 5-9. ULTIMATE FUSELAGE LOADS

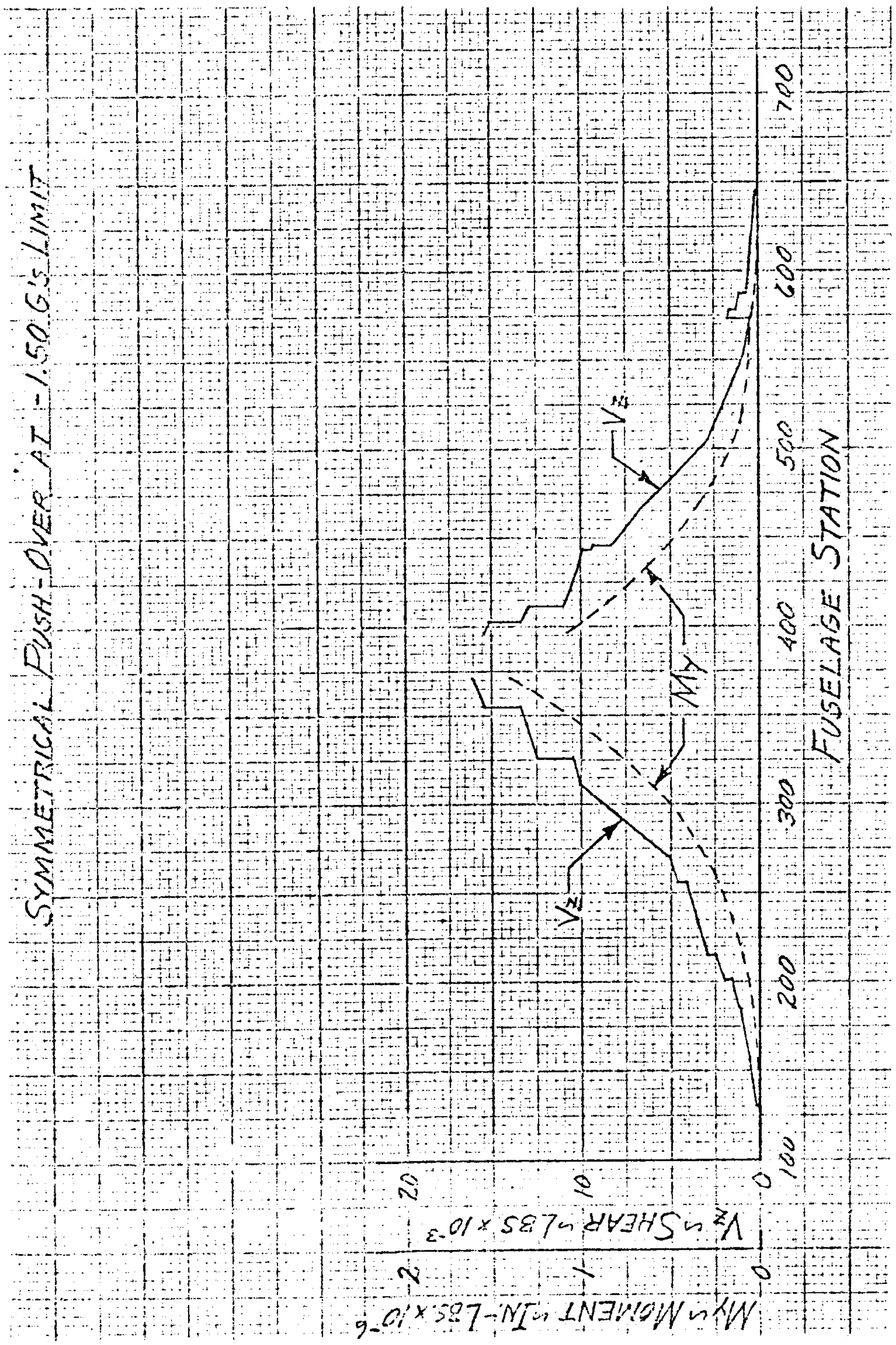

FIGURE 5-10. ULTIMATE FUSELAGE LOADS

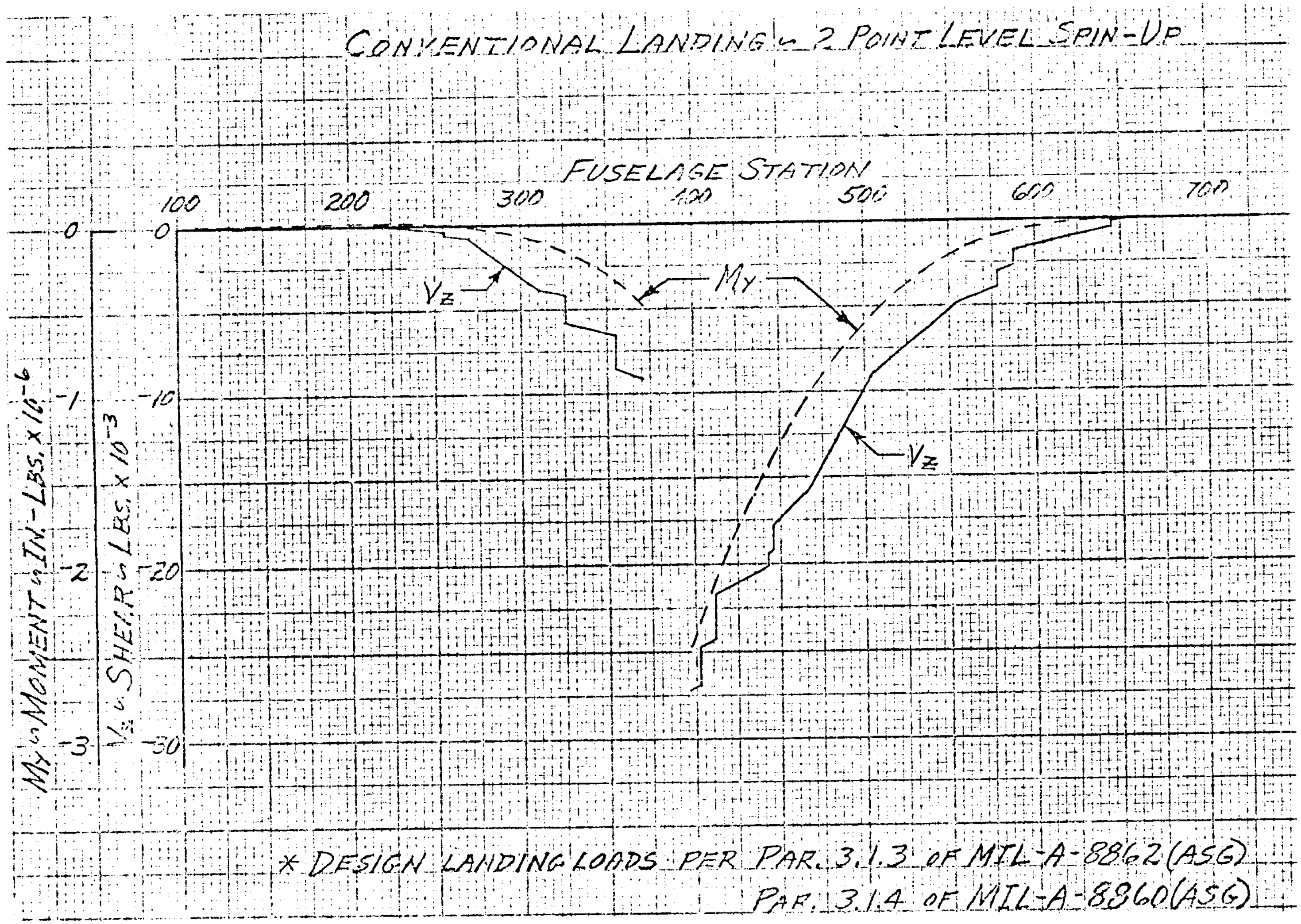

FIGURE 5-11. DESIGN FUSELAGE LOADS*

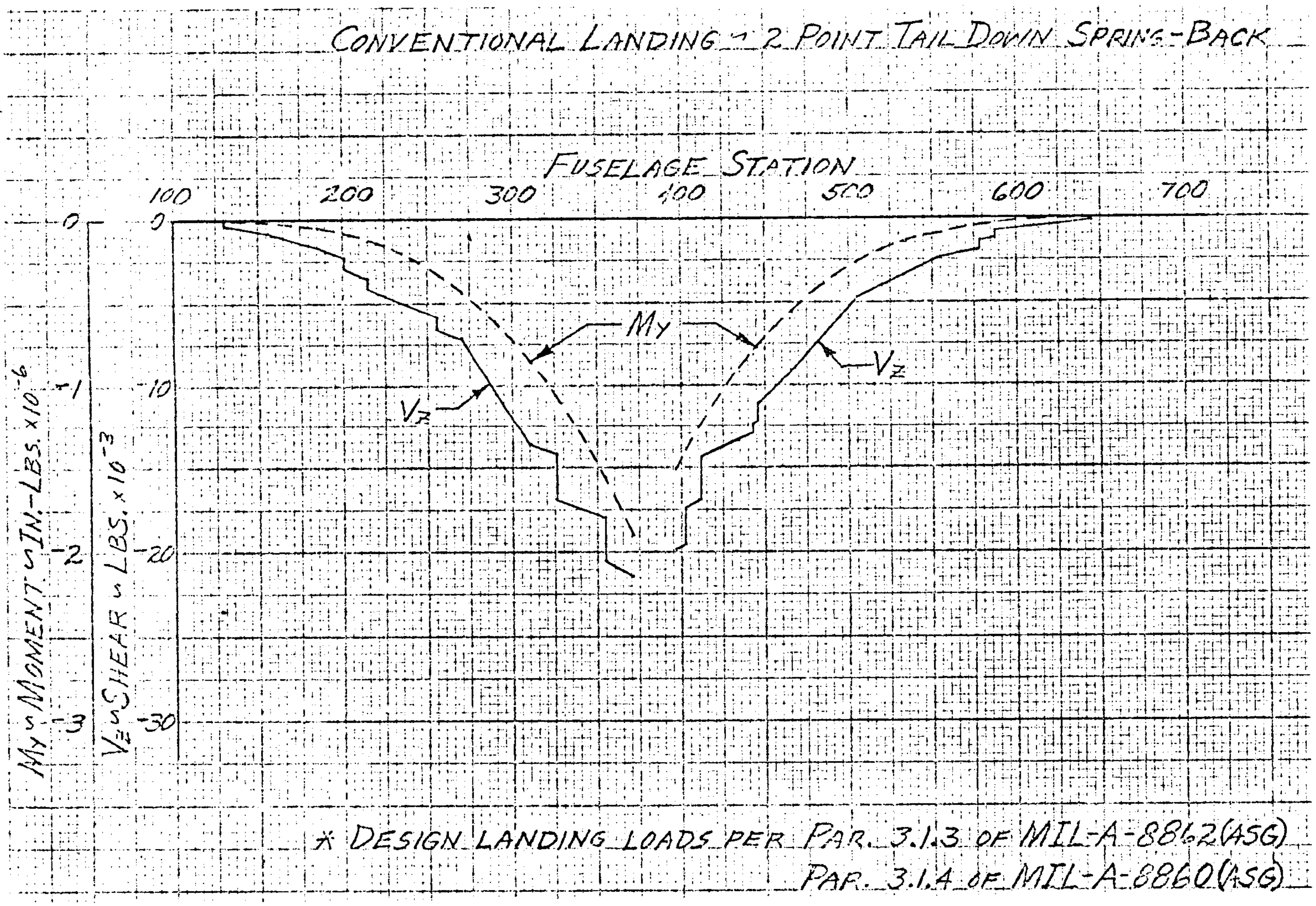

FIGURE 5-12. DESIGN FUSELAGE LOADS*

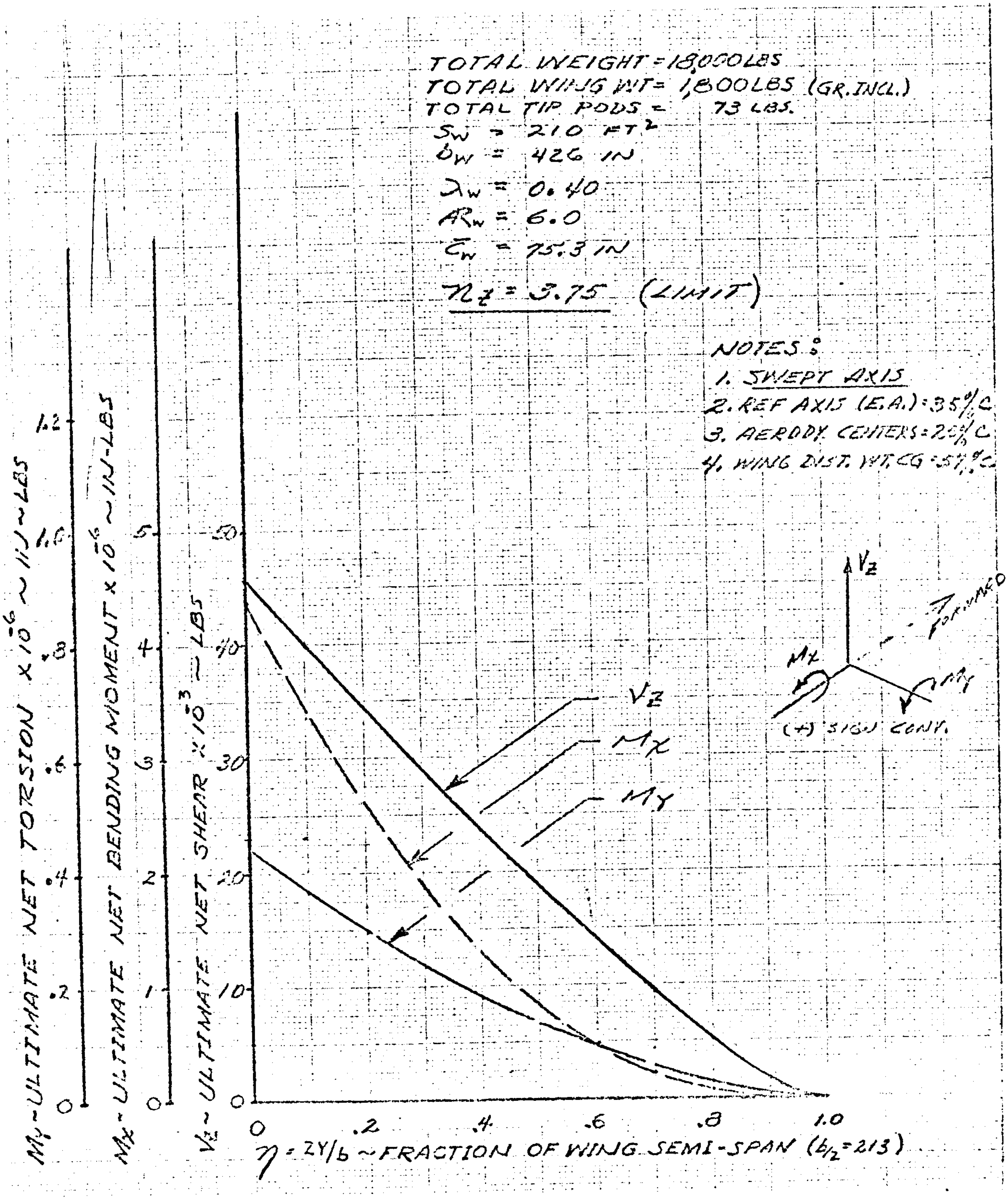

FIGURE 5-13. ULTIMATE WING SEMI-SPAN NET LOADS

horizontal tail airload. Horizontal tail airloads based on preliminary aerodynamic data are included in the aft fuselage loads. Tail loads used are 2,200 pounds up load at + 3.75 g and 880 pounds down load at - 1.5 g.

Wing loads are computed for + 3.75 g using a rigid wing and represent maximum positive bending moment on the wing. Due to the subsonic Mach range of the airplane, the wing loads are based on a chordwise center of pressure at 25 percent of the wing chord. Wing loads for - 1.5 g are not computed but would produce negative loads with a magnitude equal to approximately 40 percent of those computed for + 3.75 g.

5.5.2.2 LANDING CONDITIONS. Landing gear loads are computed for conventional landings and vertical landings. Main gear loads in vertical landing with obstruction criteria are computed.

Based on the landing conditions investigated, the two point conventional landing conditions result in the highest vertical bending moments on the fuselage. The method of analysis used is that of MIL-A-8862(ASG). Conditions for which fuselage bending moments are shown are:

1. Two point level spin-up

   η_z = 2.59 g limit $\ddot{\theta}_y$ = -4.79 Rad./Sec.^{2} limit

   η_z includes 1.59 g gear load factor plus 1.0 g due to wing lift = weight.

   $\ddot{\theta}_y$ is nose down pitch acceleration produced by main gear loads.

2. Two point tail down spring-back

   η_z = 2.81 g limit $\ddot{\theta}_y$ = +0.64 Rad./Sec.^{2} limit

   η_z includes 1.81 g gear load factor plus 1.0 g due to wing lift = weight.

   $\ddot{\theta}_y$ is nose up pitch acceleration produced by main gear loads.

5.5.2.3 FORCES DUE TO ATTITUDE CONTROL JETS. The following forces and locations were used:

Forward fuselage at F.S. 121.8	1460 pounds up
Aft fuselage at F.S. 649.8	1460 pounds up
Wing tips at W.S. 228	390 pounds up or 324 pounds down

Forces due to control jets generally will not be critical on primary structure except in the local areas of the installations.

5.5.2.4 GUST LOADS ON EXPOSED TAIL SURFACES. Loads due to gust are based on a equivalent gust velocity (U_{d_e}) of 50 fps and a flight condition of 0.75 Mach, Ve = 450 Knots, q = 685 psf.

Estimated lift curve slopes for the exposed surfaces at 0.75 Mach are:

Horizontal Tail: $C_{1\alpha}$ = 0.30 per Rad. Total for Both Surfaces

Vertical Tail: $C_{1\alpha}$ = 0.65 > per Rad.

Reference Area: = Swing = 210 Ft.^{2}

Gust loads on exposed tail surfaces due to 50 fps gust

Horizontal Tail: 1,285 pounds limit per side.

Vertical Tail: 6,230 pounds limit

Loads assumed to act at 25 percent MAC of each surface.

5.5.2.5 DESIGN LANDING GEAR LOADS.

5.5.2.5.1 Conventional Landing at 12 FPS Sink Speed.

Condition	Gear	Parallel to Oleo Lbs.	Normal to Oleo Lbs.	Load Point
Three Point Level	Nose			
Spin-up		4,790	3,690	Axle
Spring-back		4,790	-3,290	
Max. Vert. Reaction		4,790	1,200	
Two Point Level	Main			
Spin-up		14,510	11,960	Axle
Spring-back		15,080	-10,230	
Max. Vert. Reaction		14,970	4,210	
Two Point Tail Down	Main			
Spin-up		16,100	7,680	Axle
Spring-back		15,050	-11,840	
Max. Vert. Reaction		15,715	1,110	

5.5.2.5.2 Vertical Landing at 15 FPS Sink Speed.

Two Point Level - Main Gear

22,700 pounds acting normal to the ground acting alone and in combination with a horizontal load of 11,350 pounds acting forward, aft, inboard, or outboard.

Nose Down - Nose Gear

11,500 pounds acting normal to the ground acting alone and in combination with a horizontal load of 8,050 pounds acting forward, aft, inboard, or outboard.

NOTE: Loads are positive up and aft.

5.5.3 Landing Gear and Landing Flap Extension

Maximum speed with landing gear or landing flaps in extended position is 220 knots.

5.5.4 Altitude Versus Allowable Mach for Gust Velocity (U_{d_e}) = 50 FPS and Total Load Factor = 3.0 g

Allowable Mach numbers are shown over a range of altitudes for airplane gross weights from take off weight to empty weight (refer to Figure 5-14). These data are based on a gust velocity (U_{d_e}) of 50 fps and an incremental load factor due to gust of 2.0 g. The curves show that at any given altitude the allowable Mach number decreases as the airplane weight decreases and for any given weight the allowable Mach number increases as the altitude increases. It should be noted that the take off weight and empty weight are unrealistic at altitude but are shown on the curves to permit interpolation.

5.5.5 Altitude Versus Allowable Gust Velocity (U_{d_e}) for Total Load Factor = 3.0 g at Maximum Speed of Ve = 450 Knots or 0.80 Mach, Whichever is Lower

Using the speed envelope proposed by NASA of 450 KNOTS EAS (q = 685 PSF) OR 0.80 MACH, the curves indicate the gust velocity which will result in an incremental load factor of 2.0 g at each altitude and airplane gross weights ranging from

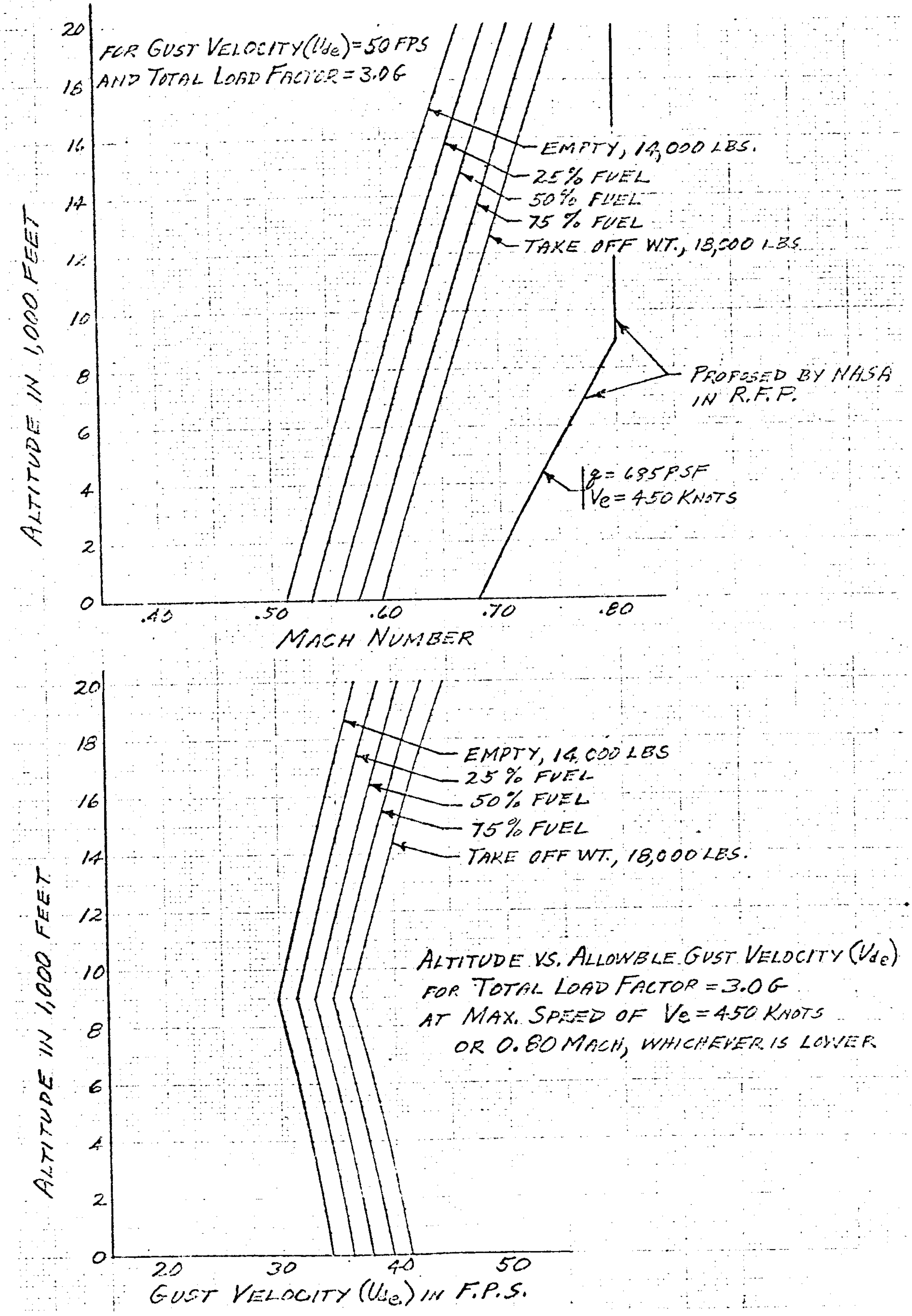

FIGURE 5-14. ALTITUDE VS. ALLOWABLE MACH

take off weight to empty. The data show that the allowable gust velocity decreases with decrease in airplane weight. The allowable gust velocity also decreases with altitude from sea level to 9,000 feet and then increases with altitude above 9,000 feet. Above 9,000 feet the maximum speed is 0.80 Mach while below 9,000 feet it is 450 KEAS based on a dynamic pressure of 685 PSF.

5.6 BASIC LOADS - MODIFIED T-39A

5.6.1 Discussion of Loads

Operating limitations will be established to assure that design limit loads will not be exceeded on the existing T-39A structure retained for the V/STOL configuration (refer to Figure 5-15). Available data indicate that the T-39A wing structure has the capability of approximately + 3.3 g in the V/STOL configuration for positive symmetrical maneuvers. Therefore, an operating limit of + 3.0 g is realistic. New structure, which will not be static tested, will be designed to loads based on 1.25 times the operating limits to assure that loads experienced in actual operation do not exceed 80 percent of design limit loads.

The allowable sink speeds versus aircraft weight presented in Figure 5-16 are based on the landing gear proposed for the installation. Analysis of the backup structure and possible beef-up are required. In general, considerable more data on the T-39A structure and design loads are required in order to properly evaluate the capability of the existing structure for the V/STOL configuration.

5.7 FLUTTER STABILITY - MODEL N-309

Flutter analyses of the wing, horizontal tail, and vertical tail of the Model N-309 airplane show freedom from flutter* throughout the airplane speed range (to

*With the stipulation that the angular flexibility from the root of the horizontal tail surface through the torque-tube and actuator must not exceed 64 x 10^{-8} radians/lb-in per side, the value used in Case 60 flutter analysis shown in Figure 5-18. This value is achieved using the X-21 aileron actuator (with F-5 horizontal actuator linear stiffness) on an eight-inch horn radius with the present torque tube design of five inches between actuator and surface root.

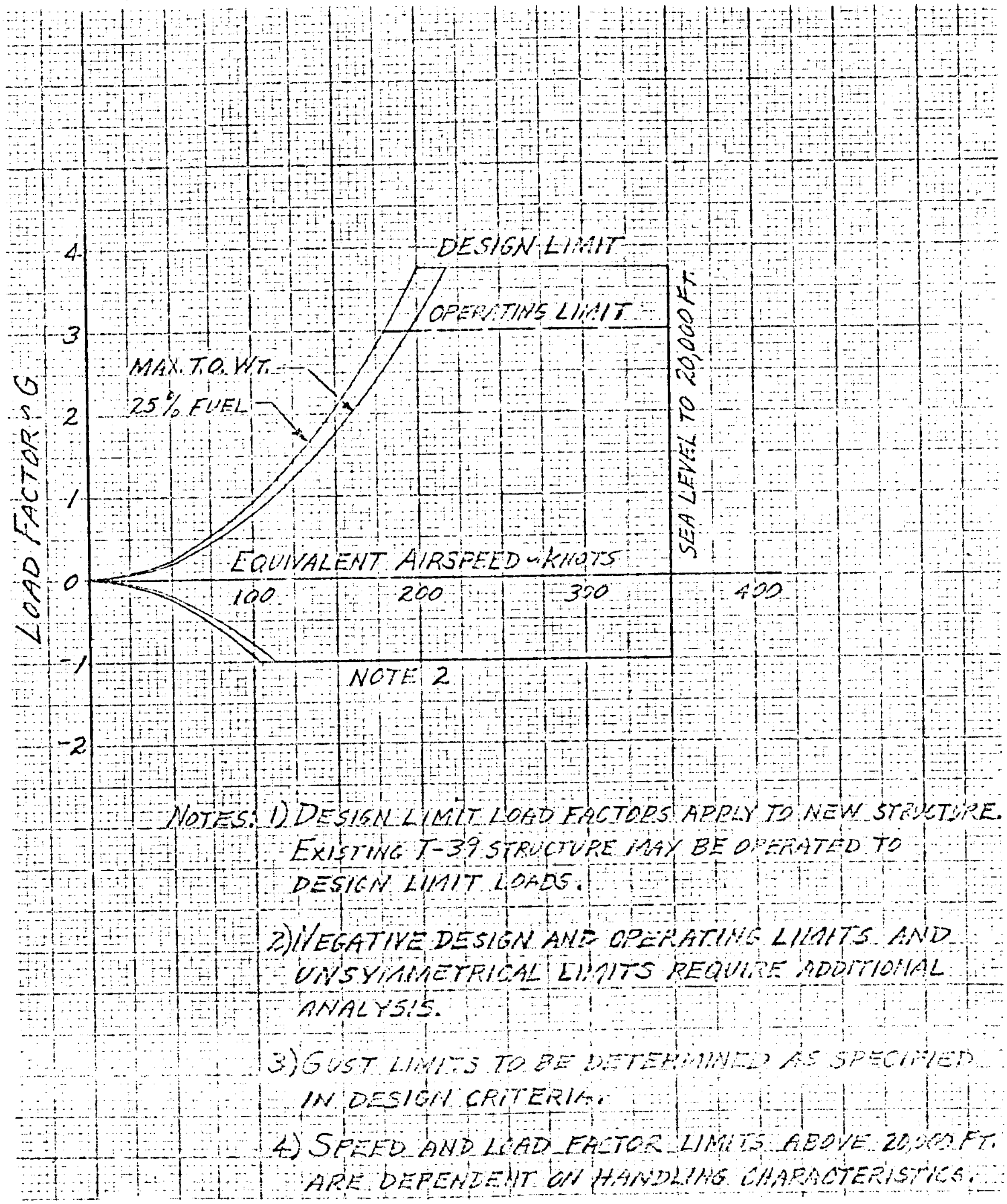

FIGURE 5-15. DESIGN AND OPERATING LIMITS FOR SYMMETRICAL MANEUVER

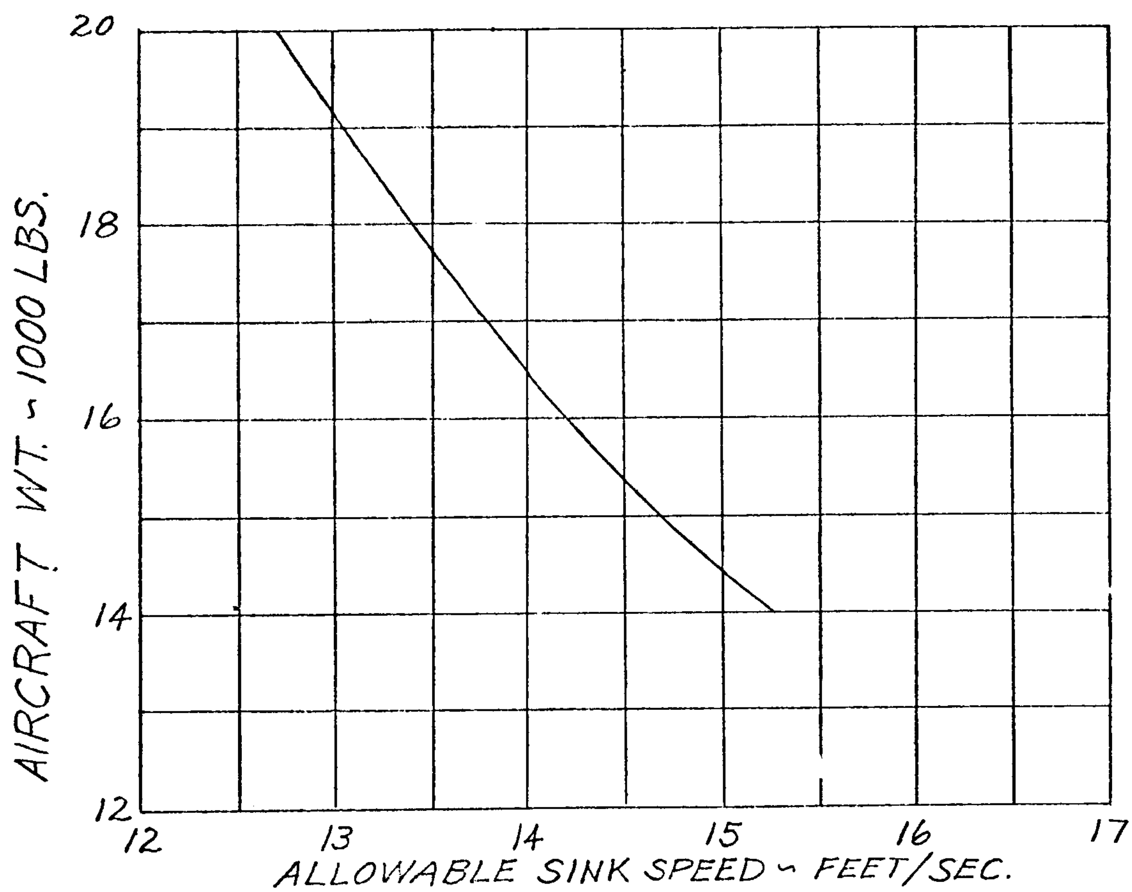

FIGURE 5-16. TWO POINT VERTICAL LANDING – MAIN GEAR
2.5 G INCREMENTAL LOAD FACTOR

Mach 0.75), including a 15 percent margin in equivalent airspeed obtained by analysis at - 8,000 feet standard altitude. The analysis results are shown in Figures 5-17 through 5-19 as plots of necessary damping, g, versus knots equivalent airspeed, KEAS. Stability is indicated by a "g" value less than 0.02. Computed frequencies in cps are shown by each plotted point.

The flutter analyses were performed in local coordinates, using subsonic, lifting surface aerodynamic force coefficients, computed for 0.75 Mach number. A seven-strip wing and five-strip tails were used. The wing and horizontal tail were connected to a rigid fuselage in free-free symmetric analyses; the vertical tail was connected to a rigid base in a cantilevered analysis.

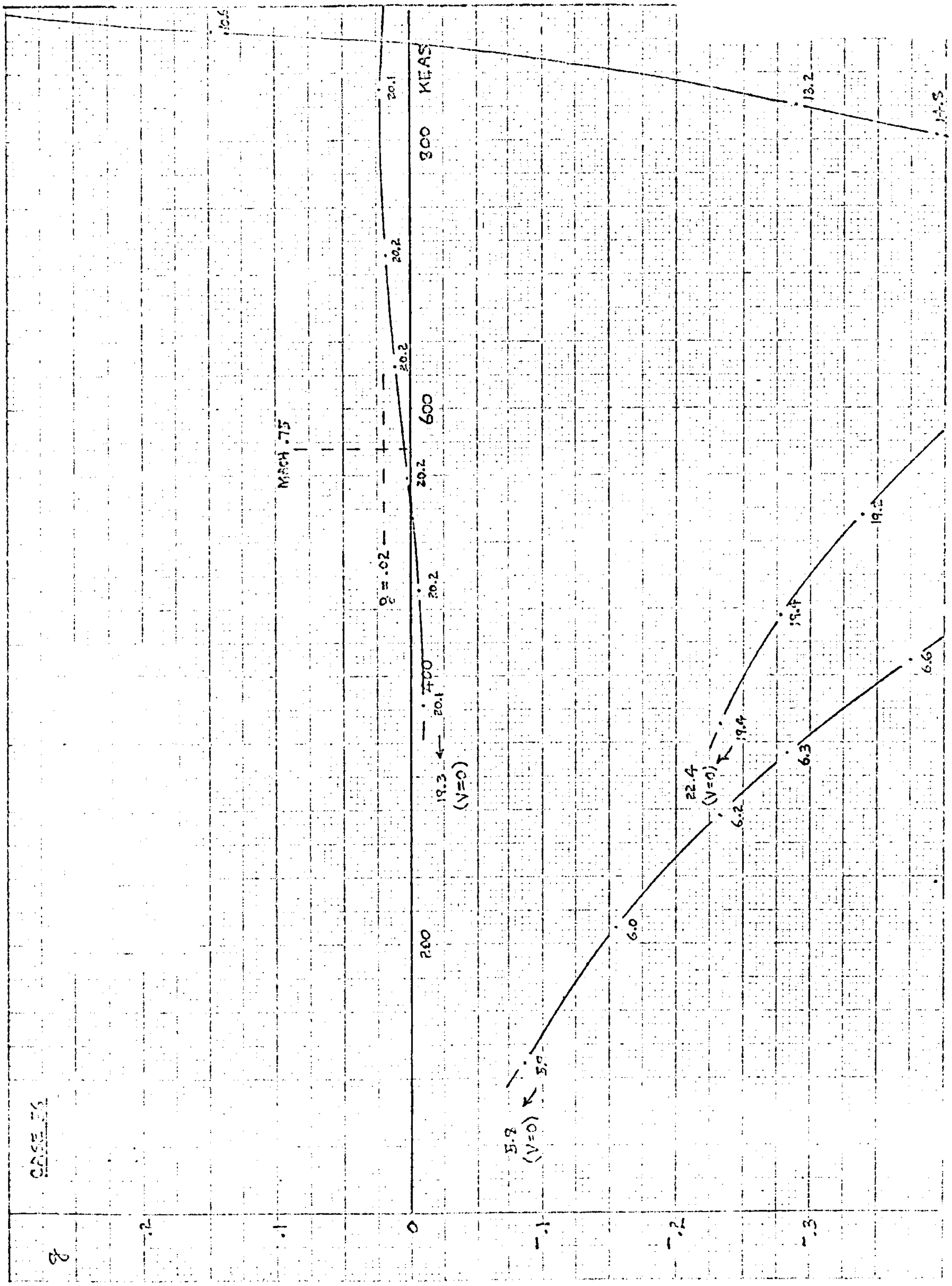

FIGURE 5-17. N-309 WING FLUTTER STABILITY CURVES ALTITUDE - 8000 FEET MACH .75

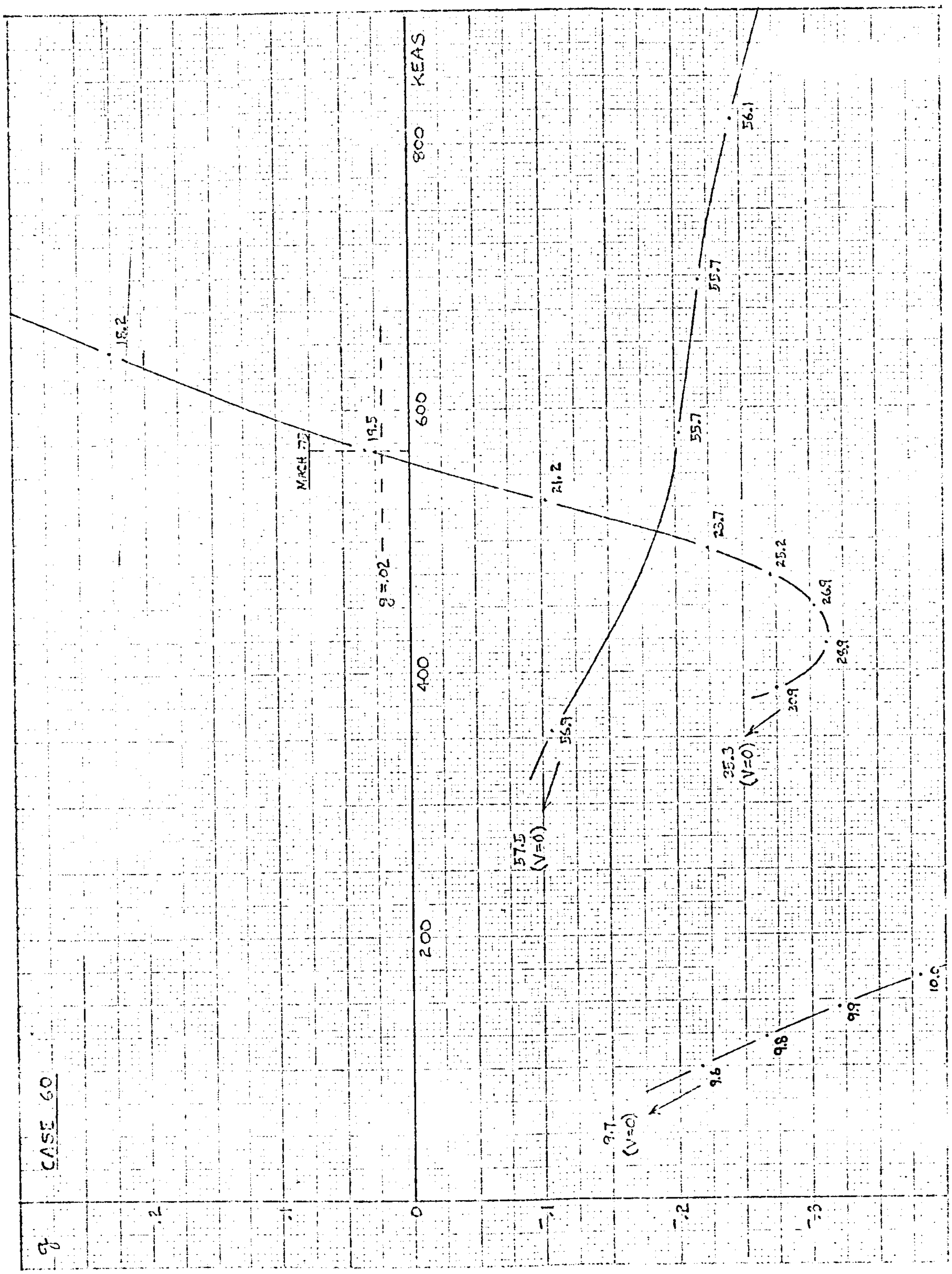

FIGURE 5-18. N-309 HORIZONTAL TAIL FLUTTER STABILITY CURVES
ALTITUDE - 8000 FEET MACH .75

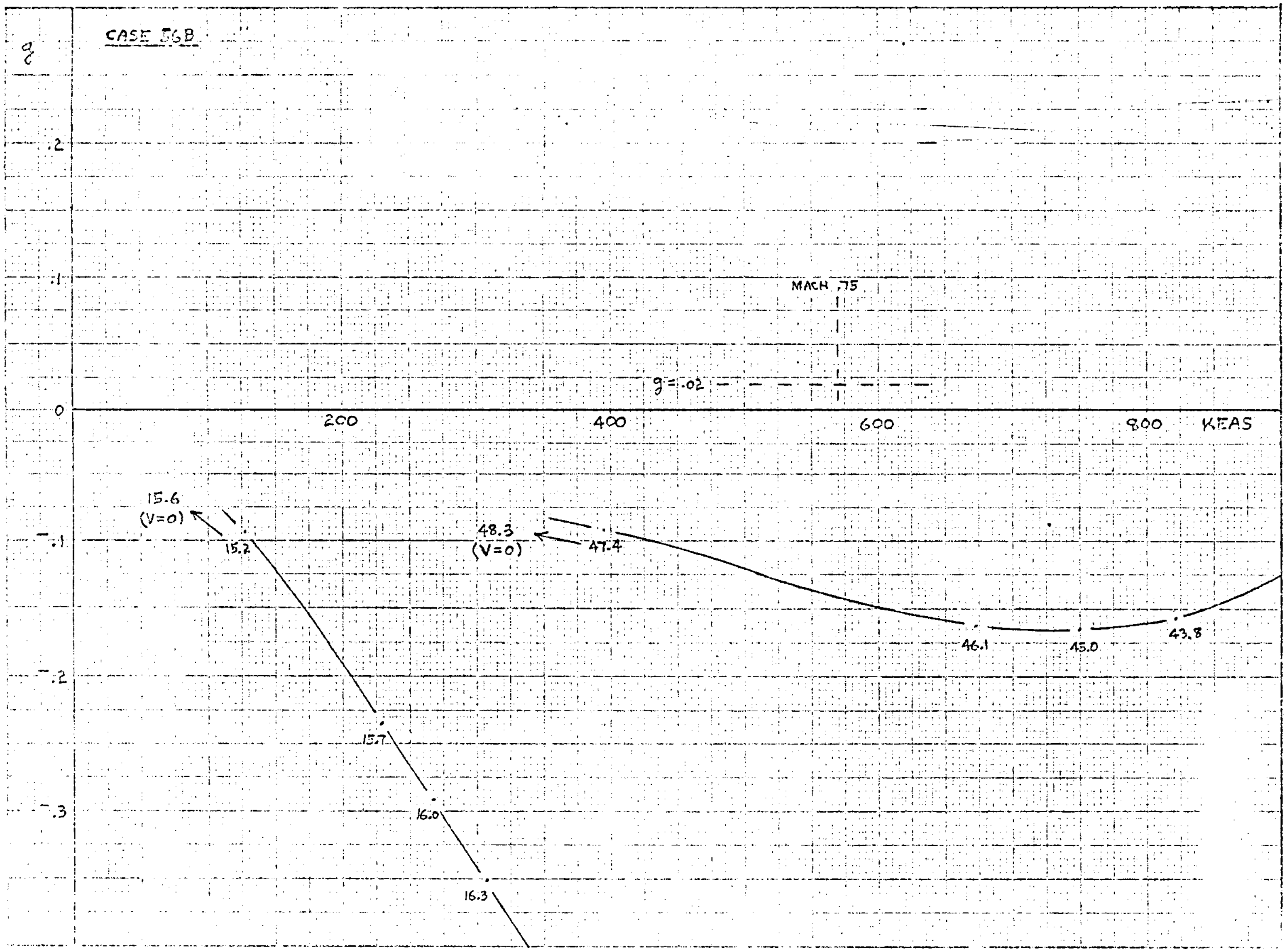

FIGURE 5-19. N-309 VERTICAL TAIL FLUTTER STABILITY CURVES
ALTITUDE – 8000 FEET MACH .75

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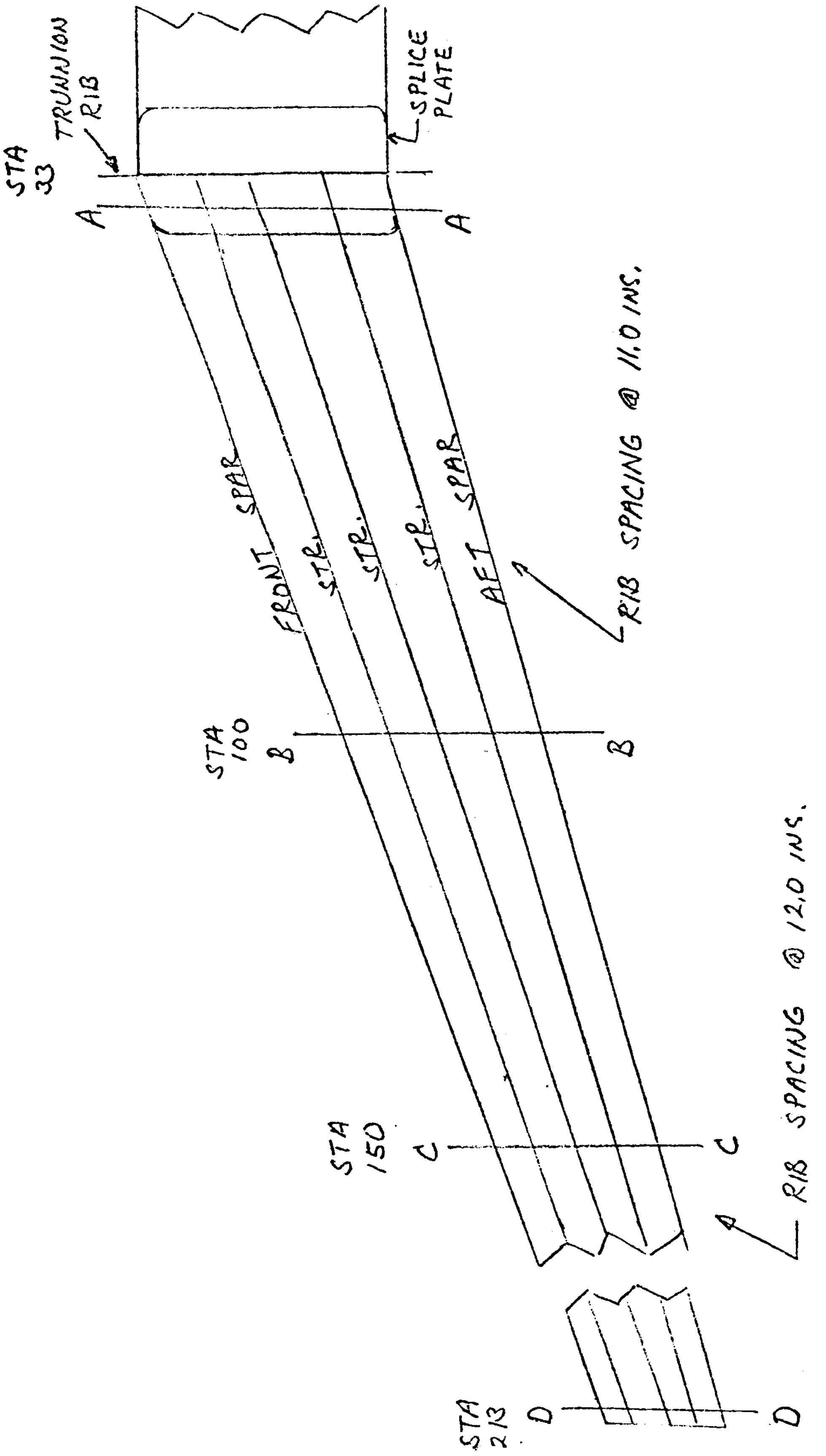
APPENDIX 5-I

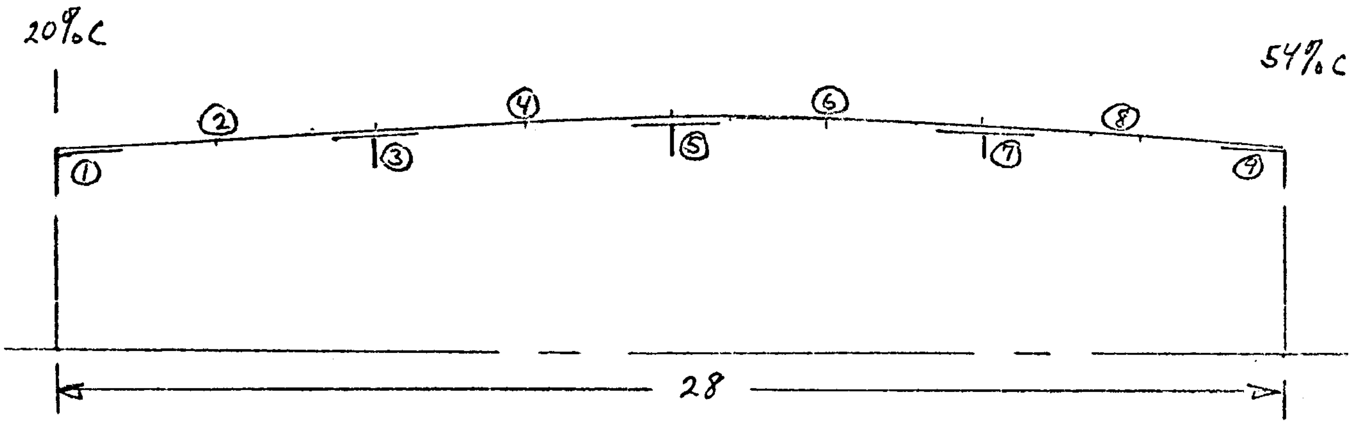
PRELIMINARY STRESS ANALYSIS


STA 33
TRUNNION RIB
SPLICE PLATE
A
A
FRONT SPAR
STR.
STR.
STR.
AFT SPAR
RIB SPACING @ 11.0 INS.
STA 100
B
B
RIB SPACING @ 12.0 INS.
STA 150
C
C
STA 213
D
D

WING

STA. 34 (OUTBD.)

BENDING SECTION NORMAL TO .35C (E.A.)



ITEM	A	Y	AY	AY^2
1	.55	5.0	2.75	13.75
2	1.05	5.3	5.55	29.50
3	.55	5.3	2.91	15.45
4	1.05	5.8	6.09	35.30
5	.55	5.9	3.24	19.15
6	1.05	6.0	6.30	37.80
7	.55	5.7	3.14	17.85
8	1.05	5.6	5.89	33.00
9	.55	5.2	2.86	14.87
1′	.55	-5.0	-2.75	13.75
2′	.63	-5.3	-3.34	17.70
3′	.55	-5.3	-2.91	15.45
4′	.63	-5.8	-3.65	21.20
5′	.55	-5.9	-3.24	19.15
6′	.63	-6.0	-3.78	22.65
7′	.55	-5.7	-3.14	17.85
8′	.63	-5.6	-3.53	19.75
9′	.55	-5.2	-2.86	14.87
Σ	12.22	$\bar{Y}$ = .78	9.53	379.04

UPPER SURFACE (items 1–9)

LOWER SURFACE (items 1′–9′)

$t_{UPP.\ SKIN} = .150$

$t_{LWR.}$ " $= .090$

$$J = \frac{(4)(310^2)}{(28/.15 + 28/.09 + 20/.10)}$$

$= 550\ IN^4$

$GI = (3.9 \times 10^6)(550)$

$= 1,520 \times 10^6\ LB.IN^2$

$EI = (10.3 \times 10^6)(372.6)$

$= 3,840 \times 10^6\ LB.IN^2$

$I = 379.04 - 12.22 \times .78^2$

$= 372.6\ IN^4$

STA. 34 (OUTBD.)

M_x = 3,000,000 IN. LBS. ULT
M_y = 330,000 IN. LBS. "
V_z = 37,000 LBS. ULT.

$$f_{b_c} = (3,000,000)(5.22)/372.6 = 42,000 \text{ psi}$$

COLUMN ALLOW. STR. (EXTR. AND 10136-2403)
(7075-T6)

$$L/\rho = \frac{11}{.333} = 33 \qquad F_c = 55,000 \text{ psi.}$$

F_{MAX} (SKIN) = 46,000 psi.
@ 180°F

$$b/t = \frac{4.5}{.15} = 30 \qquad F_{MAX} = (.94)(46,000) = 43,000 \text{ psi}$$

$$M.S. = \frac{43}{42} - 1 = +\underline{\underline{.02}}$$

$$f_{b_t} = (3,000,000)(6.78)/372.6 = 54,500 \text{ psi}$$

F_{tu} = 76,000 psi.
(7075-T6)

$$M.S. = \frac{(.94)(76)}{54.5} - 1 = +\underline{\underline{.31}}$$

$$q_T = \frac{T}{2A} = \frac{333,000}{620} = 540 \text{ LBS/IN.}$$

N.C.

$$q_s = \frac{37,000}{(2)(10)} = 1850 \text{ LBS/IN.}$$

$$t_{REQ'D} = .100 \text{ INS.}$$

STA. 34 (SPLICE)

AREA UPPER SKIN & STR.
$= (.55)(5) + (1.05)(4) = 6.95 \text{ IN}^2$

AXIAL LOAD $(6.95)(42,000) = 292,000$ LBS

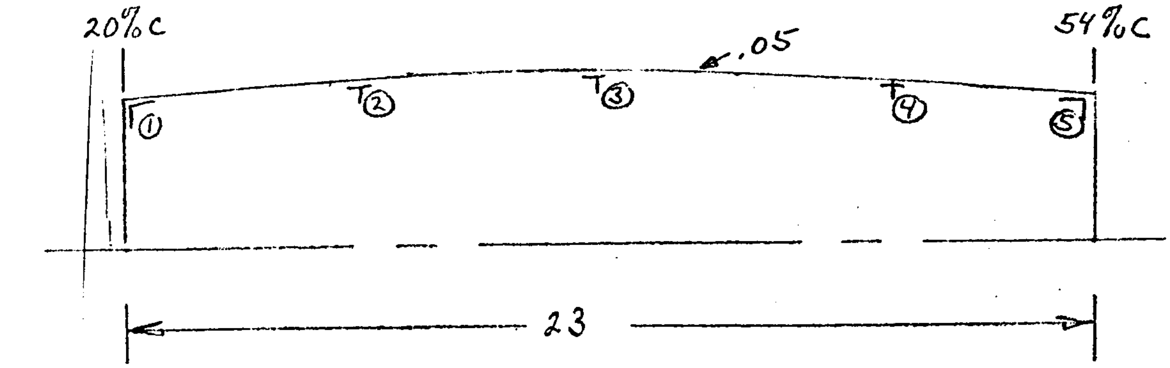
$$\frac{292,000}{(28)(t_{REQD.})} = 50,000 \text{ psi}$$

UPPER — $t_{REQ'D.} = .208$ — $t = \underline{.21}$ INS.

$$\frac{292,000}{(28)(t_{REQD.})/1.25} = 65,000$$

LOWER — $t_{REQ'D.} = .20$ $t = \underline{.20}$ INS.

STA. 100 (B-B) (BENDING SECTION NORMAL TO .35C)



(7075-T6)

	ITEM	A	Y	AY	AY^2
UPP. SURFACE	1	.850	4.0	3.40	13.60
	2	.325	4.2	1.36	5.74
	3	.325	4.5	1.46	6.60
	4	.325	4.2	1.36	5.74
	5	.850	3.8	3.23	12.30
LWR. SURFACE	1'	.850	-4.0	-3.40	13.60
	2'	.325	-4.2	-1.36	5.74
	3'	.325	-4.5	-1.46	6.60
	4'	.325	-4.2	-1.36	5.74
	5'	.850	-3.8	-3.23	12.30
	Σ	4.153			87.96

$$J = \frac{(4)(\overline{193})^2}{46/.05 + 16/.20}$$

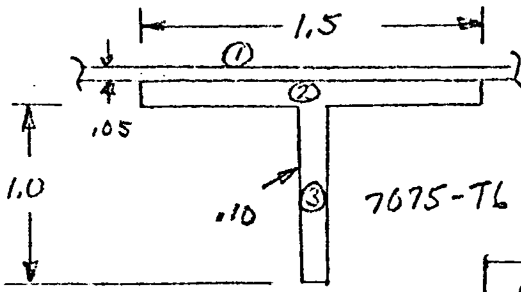
$$= 149 \text{ IN.}^4$$

$$EI = (10.3 \times 10^6)(87.96) = 905 \times 10^6 \text{ LB. IN}^2.$$

$$GJ = (3.9 \times 10^6)(149) = 581 \times 10^6 \text{ LB. IN}^2.$$

STA. 100 (B-B) (CONT.)

STRINGER



ITEM	A	Y	AY	AY^2	I_o
1	.075	.025	.0018	—	—
2	.150	.100	.0150	.0015	—
3	.100	.650	.0650	.0423	.0083
Σ	.325	$\bar{Y}$=.252	.0818	.0438	.0083

$I = .0438 + .0083 - .325 \times .252^2 = .0315 \ IN^4$

$\rho = \sqrt{I/A} = .312$

$L/\rho = 11/.312 = 35$ $\quad F_c = 54{,}000$ psi.

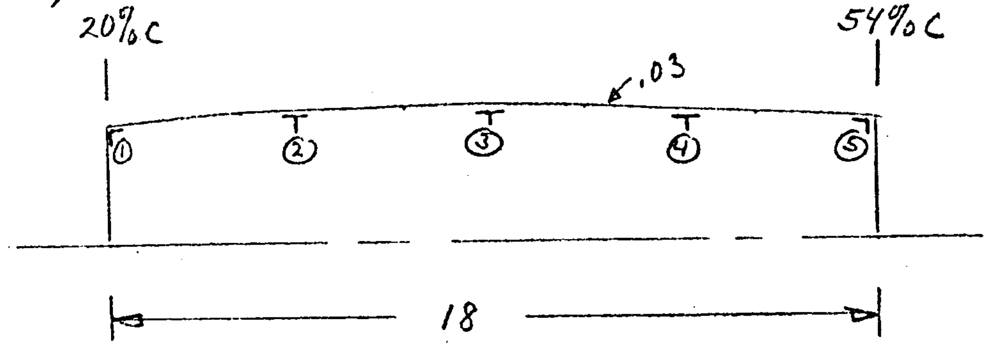
@ 180°F $\quad F_c = (.94)(54{,}000) = 50{,}700$ psi

$M_x = 950{,}000$ IN. LBS.

$f_{bc} = (950{,}000)(4.5)/87.96 = 48{,}600$ psi

$M.S. = \frac{50.7}{48.6} - 1 = +\underline{.04}$

STA. 150 (BENDING SECTION NORMAL TO .35C)
(C-C)



Ti - 6AL - 4V

UPP. SURFACE: items 1–5; LWR SURFACE: items 1'–5'

ITEM	A	Y	AY	AY^2
1	.087	2.9	.252	.732
2	.131	3.0	.393	1.180
3	.131	3.1	.406	1.260
4	.131	3.1	.406	1.260
5	.087	2.9	.252	.732
1'	.087	-2.9	-.252	.732
2'	.131	-3.0	-.393	1.180
3'	.131	-3.1	-.406	1.260
4'	.131	-3.1	-.406	1.260
5'	.087	-2.9	-.252	.732
Σ	1.134			10.33

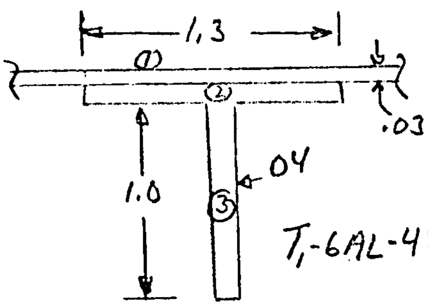
$$J = \frac{(4)(\overline{108}^2)}{47.5/.03} = 29.5 \text{ IN}^4$$

$$EI = (16 \times 10^6)(10.33) = 165 \times 10^6 \text{ LB. IN}^2$$

$$GJ = (6.2 \times 10^6)(29.5) = 183 \times 10^6 \text{ LB. IN}^2$$

STA. 150 (C-C) (CONT.)

STRINGER



T_i-6AL-4V

ITEM	A	Y	AY	AY^2	I_o
1	.039	.015	.0006	—	—
2	.052	.050	.0026	.00013	—
3	.040	.570	.0228	.01300	.0033
Σ	.131	$\bar{Y}$ = .198	.0260	.0131	.0033

$I = .0131 + .0033 - .131 \times .198^2 = .0113 \text{ IN}^4$

$\rho = \sqrt{I/A} = .294$

$L/\rho = 12.0/.294 = 40.7$

$F_c = \pi^2 E / (L/\rho)^2 = 95{,}000 \text{ psi.}$

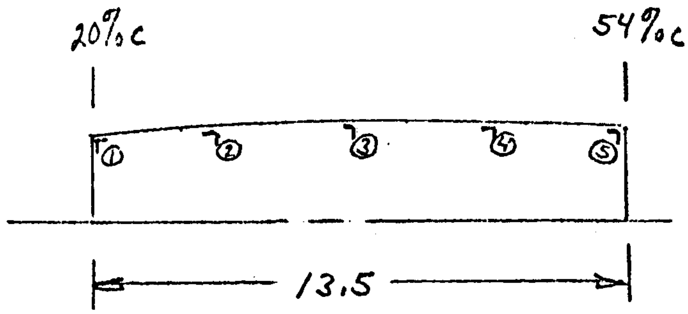
$T = 400°\,F$ $\quad F_c = (95{,}000)(.80) = 76{,}000 \text{ psi.}$

$M_X = 200{,}000$ IN. LBS.

$f_{bc} = (200{,}000)(3.1)/10.33 = 60{,}000 \text{ psi.}$

$M.S. = \frac{76}{60} - 1 = +\underline{\underline{.27}}$

STA. 213 (D-D) (BENDING SECTION NORMAL TO .35c)



MIN. GAUGES

$t_{SKIN} = .030$

$t_{CAP} = .040$

Ti-6AL-4V

	ITEM	A	Y	AY	AY^2
UPP. SURFACE	1	.087	2.2	.191	.421
	2		2.5	.218	.544
	3		2.5	.218	.544
	4		2.5	.218	.544
	5		2.4	.209	.501
LWR SURFACE	1'		-2.2	-.191	.421
	2'		-2.5	-.218	.544
	3'		-2.5	-.218	.544
	4'		-2.5	-.218	.544
	5'	.087	-2.4	-.209	.501
	Σ	.870			5.108

CAP AREA = .060 IN^2

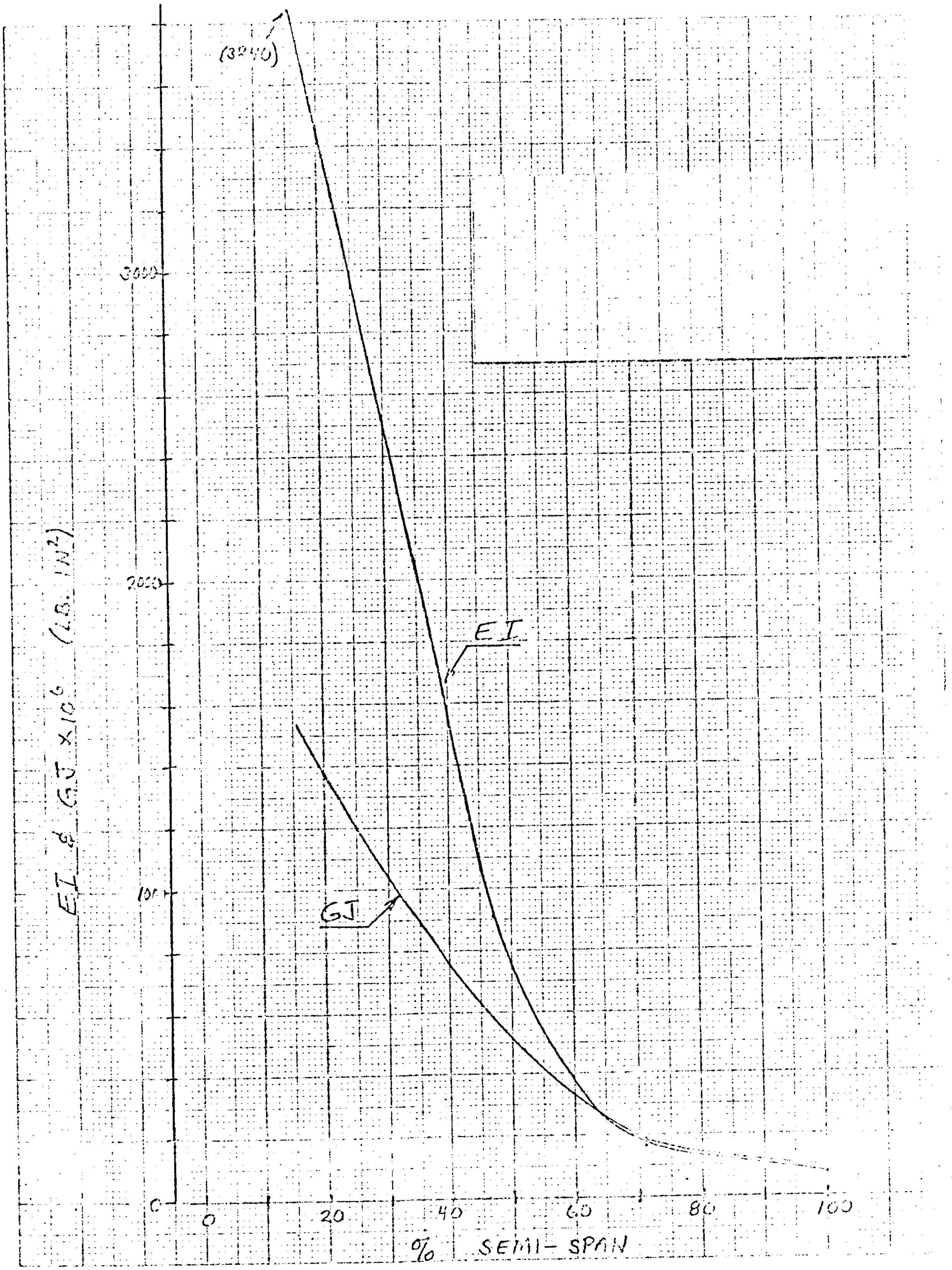
30t SKIN = .027

$\Sigma = .087 \ IN^2$

$$J = \frac{(4)(65^2)}{36/.03} = 14.1 \ IN^4$$

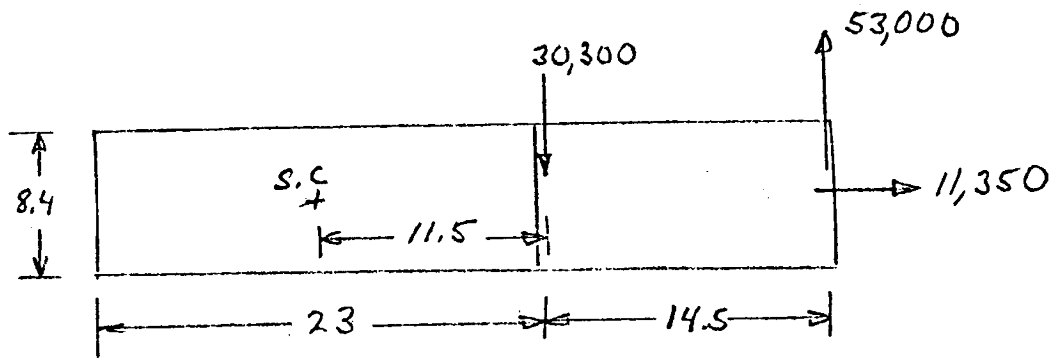
$$EI = (16 \times 10^6)(5.108) = 81.5 \times 10^6 \ LB \ IN^2$$

$$GJ = (6.2 \times 10^6)(14.1) = 87.5 \times 10^6 \ LB \ IN^2$$



WING – EI AND GJ V.S. SEMI-SPAN

RIB @ MAIN LANDING GEAR TRUNNION



LOADS @ S.C.

$T = (53{,}000)(26) - (30{,}300)(11.5) = 1{,}032{,}000$ IN. LBS.

$V = 22{,}700$ LBS.

$D = 11{,}350$ LBS.

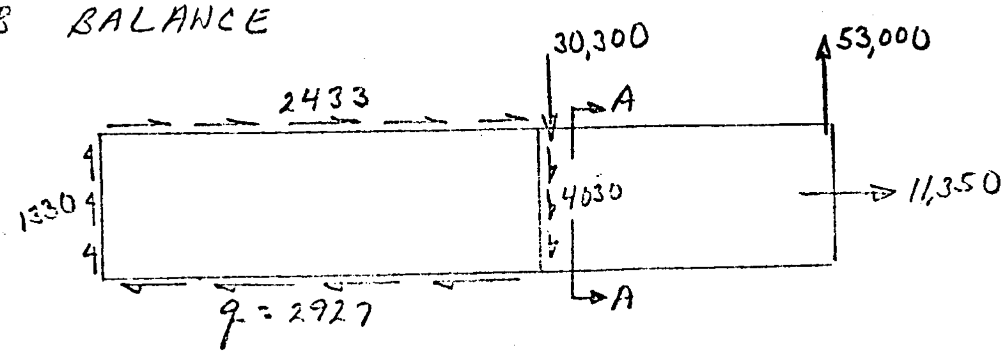
$\text{AREA} = (23)(8.4) = 193 \text{ IN}^2.$

$$q_T = \frac{1{,}032{,}000}{(2)(193)} = 2{,}680 \text{ LBS/IN.}$$

$$q_V = \frac{22{,}700}{(2)(8.4)} = 1{,}350 \text{ LBS/IN}$$

$$q_D = \frac{11{,}350}{(2)(23)} = 247 \text{ LBS. IN.}$$

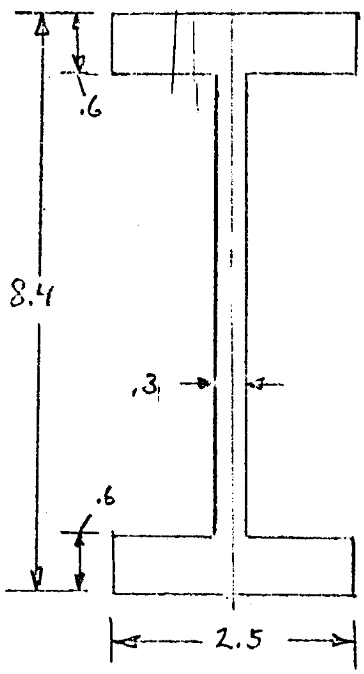
RIB BALANCE



RIB @ MAIN LANDING GEAR TRUNNION

SECTION A-A

$M = (53{,}000)(14.5) = 770{,}000$ IN. LBS.
$V = 53{,}000$ LBS
$D = 11{,}350$ LBS.



$I = 2(2.5)(.6)(\overline{3.9}^2) + \frac{1}{12}(.30)(\overline{7.2}^3) = 54.9 \ IN^4$

$A = (2)(2.5)(.6) + (.3)(7.2) = 5.16 \ IN^2$

$Q = (2.5)(.6)(3.9) + (.3)(3.6)(1.8) = 6.30 \ IN^3$

$f_{bc} = \frac{(770{,}000)(4.2)}{54.9} = 59{,}000$ psi

$f_t = 11{,}350/5.16 = 2{,}200$ psi

$\Sigma = 57{,}000$ psi

$F_{cy} = 64{,}000$ psi

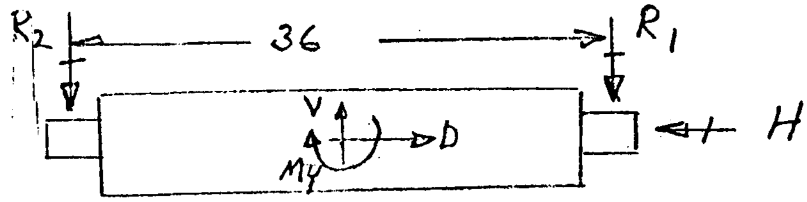
$M.S. = \frac{64}{57} - 1 = +\underline{\underline{.12}}$

$f_s = \frac{(53{,}000)(6.3)}{(.3)(54.5)} = 20{,}400$ psi

N.C.

TRUNNION RIB WING TO FUSELAGE (TRUNNION LDS)

MAIN LANDING GEAR LOADING



$M_Y = -1{,}505{,}000$ IN. LBS
$V_Z = 22{,}700$ LBS.
$D = 11{,}350$ LBS.

$$R_1 = 1{,}505{,}000/36 + 22{,}700/2 = 53{,}250 \text{ LBS.}$$

$$R_2 = -1{,}505{,}000/36 + 22{,}700/2 = -30{,}550 \text{ LBS}$$

$$H = 11{,}350 \text{ LBS.}$$

WING AIRLOADING

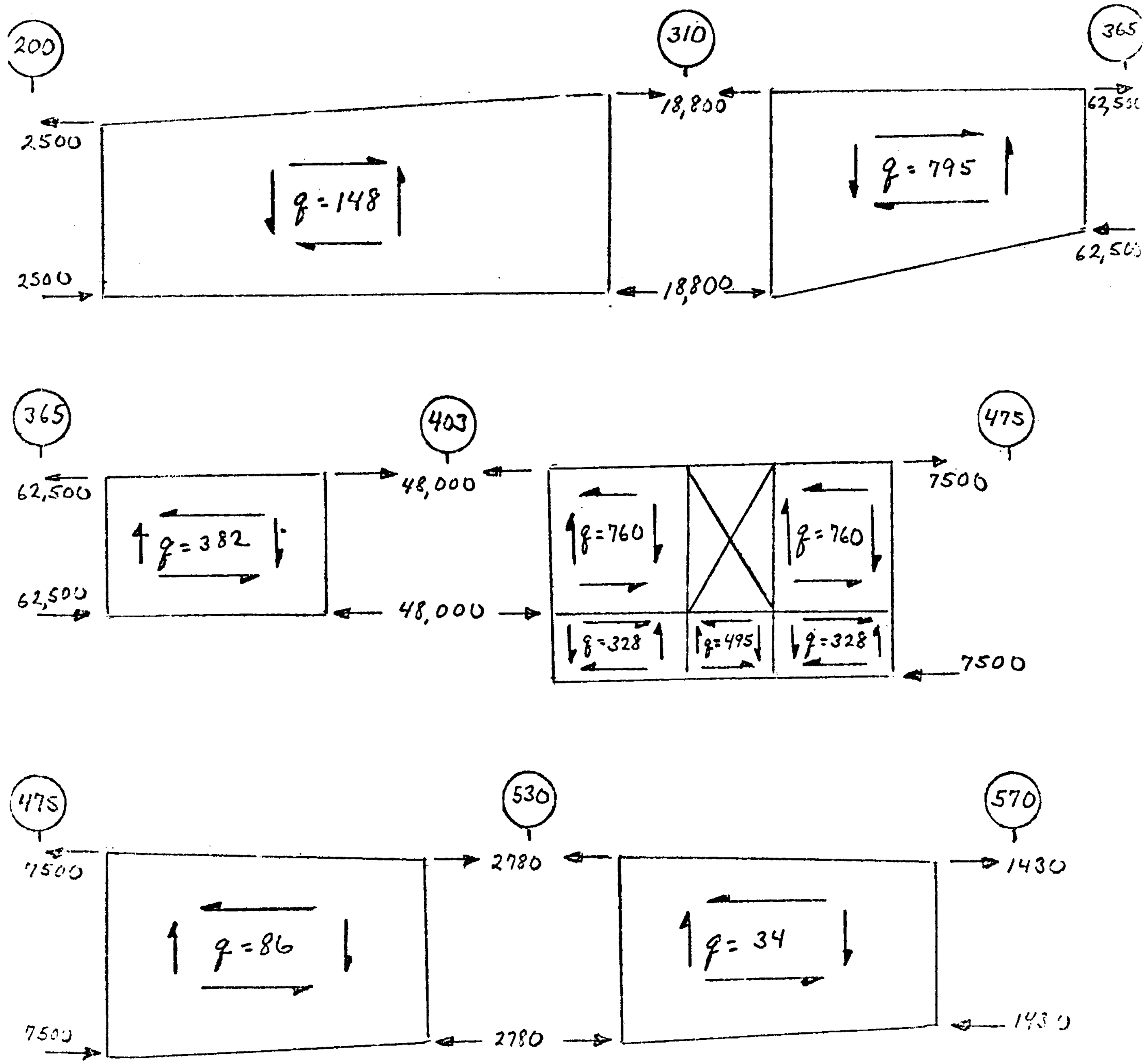
$M_Y = 330{,}000$ IN. LBS
$V_Z = 37{,}000$ LBS.

$$R_1 = -330{,}000/36 + 37{,}000/2 = 9{,}300 \text{ LBS.}$$

$$R_2 = 330{,}000/36 + 37{,}000/2 = 27{,}700 \text{ LBS.}$$

ULT. AXIAL LOADS & SHEAR FLOWS / (SIDE PANEL)

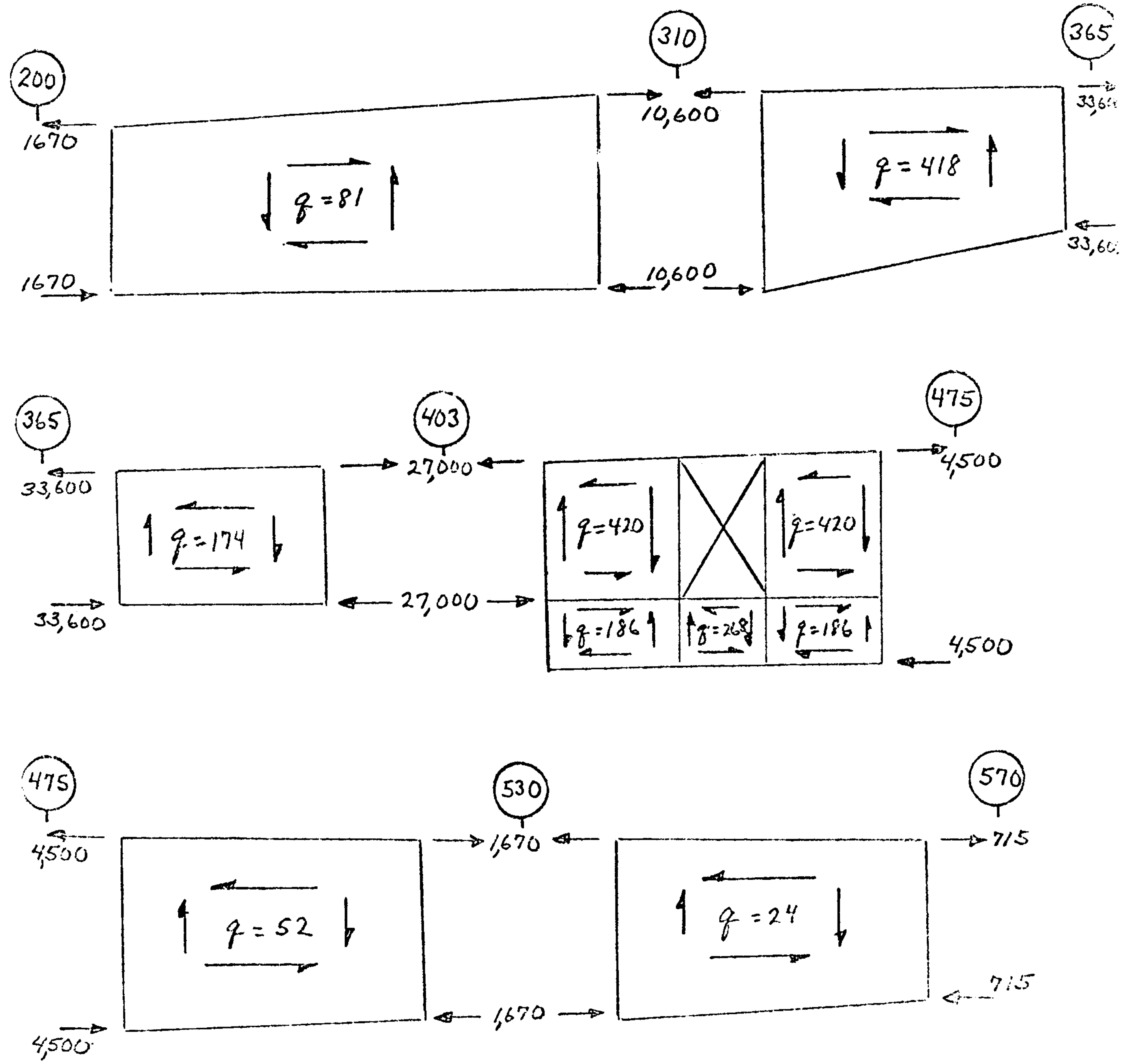
COND. SYMMETRICAL PULL-UP @ 3.75 g's LIMIT.



FUSELAGE

DESIGN AXIAL LOADS & SHEAR FLOWS (SIDE PANEL)

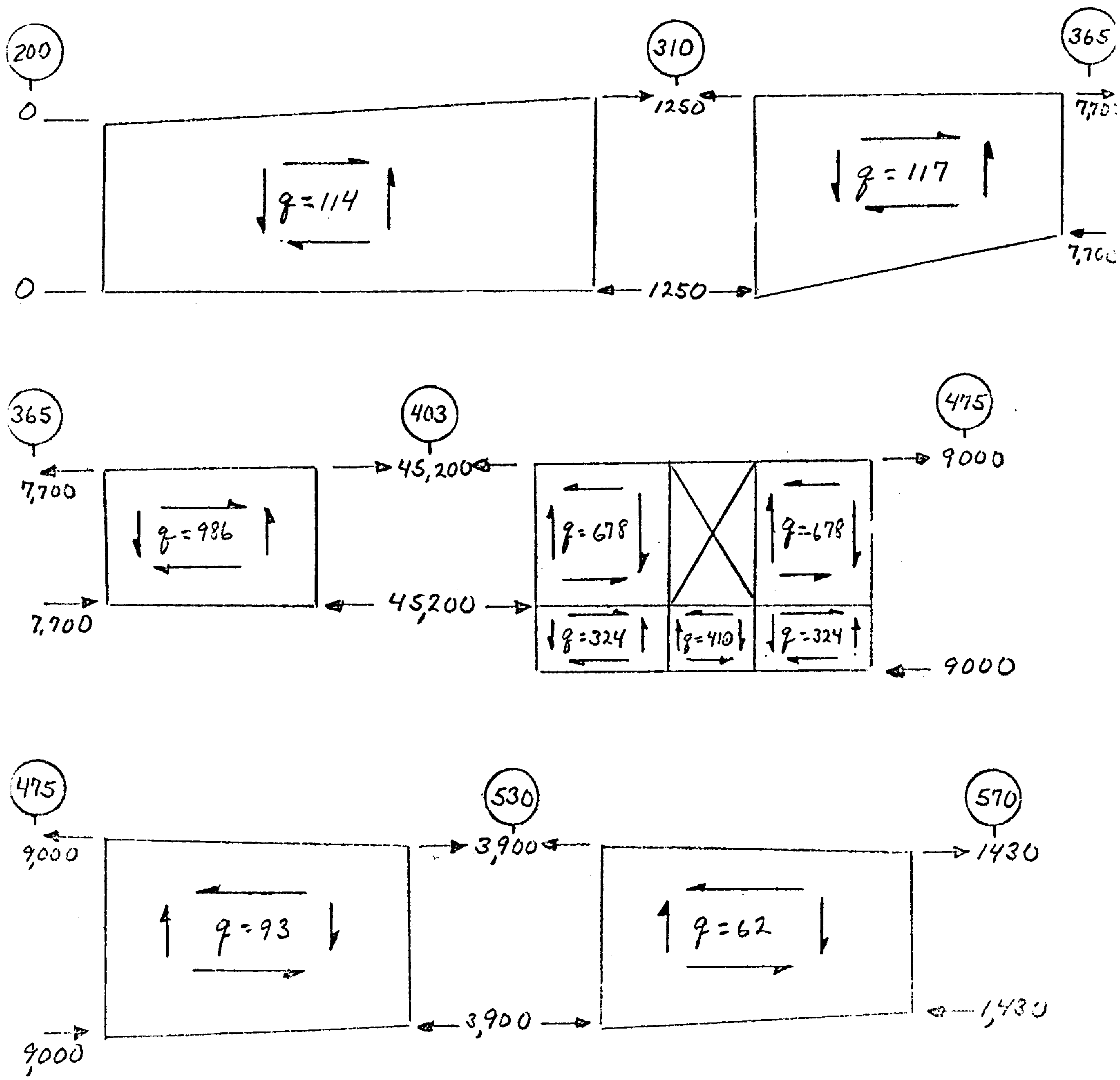
COND. CONVENTIONAL LANDING 2 PT. TAIL DOWN SPRING BACK



FUSELAGE

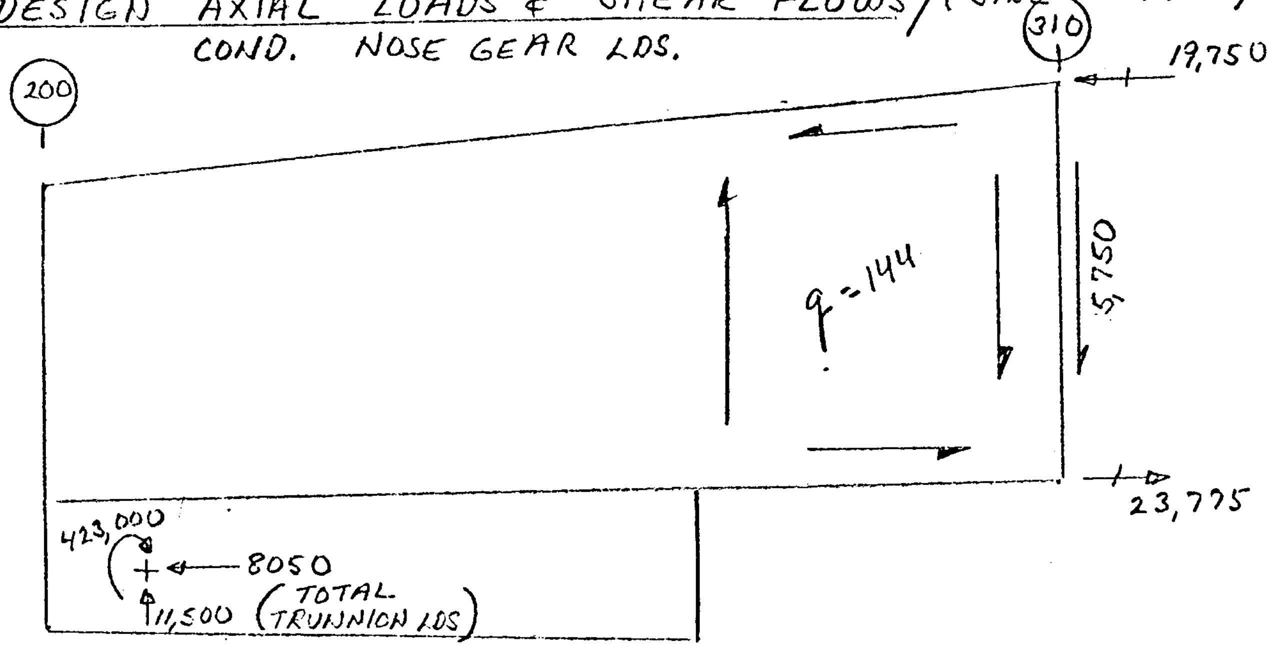
DESIGN AXIAL LOADS & SHEAR FLOWS / (SIDE PANEL)

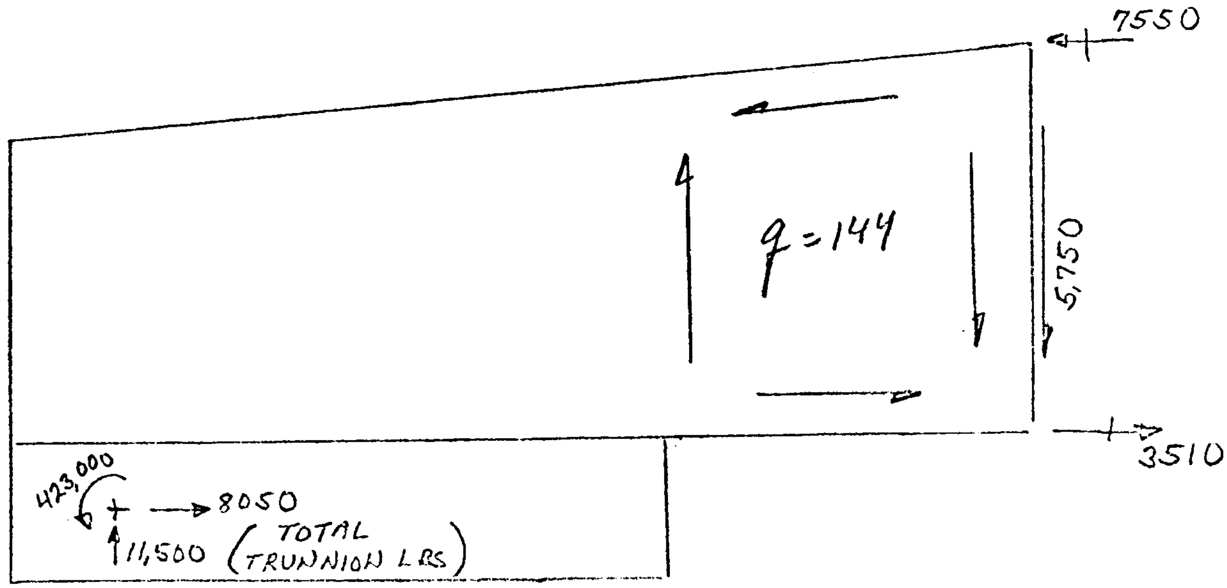
COND. CONVENTIONAL LANDING – 2 PT. LEVEL SPIN-UP



FUSELAGE

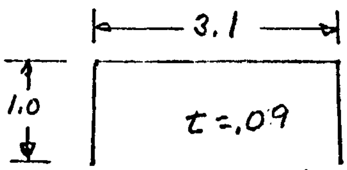
DESIGN AXIAL LOADS & SHEAR FLOWS/(SIDE PANEL)
COND. NOSE GEAR LDS.





UPPER LONGERON

STA. 310



(7075-T6)

LOAD = 19,750 LBS. DESIGN COMP.

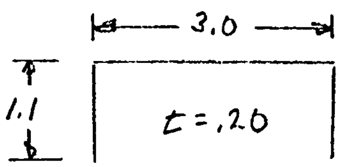
AREA = $(.09)(4.9) = .44$ IN.^{2}

$f_t = 19{,}750/.44 = 45{,}000$ psi

$F_{MAX} = 45{,}500$ psi.

$M.S. = \frac{45.5}{45.0} - 1 = +.01$

STA. 365



(7075-T6)

LOAD = 62,500 LBS. ULT. TENSION

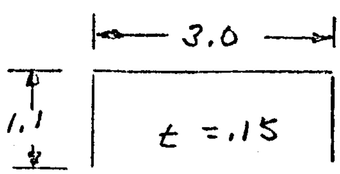
AREA = $(.20)(5.2 - .4) = .96$ IN.^{2}

$f_t = 62{,}500/.96 = 65{,}000$ psi.

$F_{tu} = 77{,}000$ psi.

$M.S. = \frac{77.0}{65.0} - 1 = +.18$

STA. 463.



(7075-T6)

LOAD = 45,200 LBS. DESIGN TENSION

AREA $= (.15)(5.2 - .3) = .735$ IN^2.

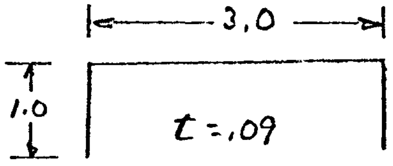
$f_t = 45{,}200/.735 = 61{,}500$ psi.

$F_{ty} = 67{,}000$ psi.

$M.S. = \frac{67.0}{61.5} - 1 = +.09$

LOWER LONGERON

STA. 310 LOAD = 18,800 LBS. ULT. COMPRESSION



(7075-T6)

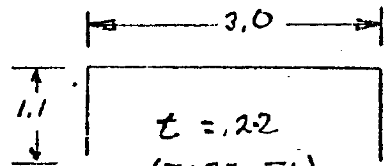
AREA = (.09)(4.8) = .433 IN.^{2}

$f_c = 18{,}800/.433 = 43{,}500$ psi

$F_{MAX} = 45{,}500$ psi

$M.S. = \frac{45.5}{43.5} - 1 = \underline{+.05}$

STA. 365 LOAD = 62,500 LBS. ULT. COMPRESSION



(7075-T6)

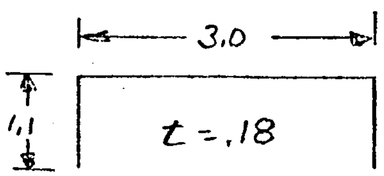
AREA = (.22)(5.2 - .44) = 1.05 IN.^{2}

$f_c = 62{,}500/1.05 = 59{,}500$ psi

$F_{MAX} = 64{,}000$ psi

$M.S. = \frac{64.0}{59.5} - 1 = \underline{+.08}$

STA. 403 LOAD = 48,000 LBS. ULT. COMPRESSION
" = 45,200 LBS. DESIGN COMPRESSION



(7075-T6)

AREA = (.18)(5.2 - .36) = .870 IN.^{2}

$f_c = 48{,}000/.870 = 55{,}000$ psi

$F_{CR} = F_{MAX}$
$= 64{,}000$ psi

$M.S. = \frac{64}{55} - 1 = \underline{+.16}$

SIDE SHEAR PANELS.

STA. 200 to 310 $\quad q = 148$ LBS/IN $\quad t = .025$ INS. (MIN.)

STA. 310 to 365 $\quad q = 795$ LBS/IN. $\quad t = .035$ INS

$d/t = \frac{5.0}{.035} = 143 \quad h/t = 40/.035 = 1140 \quad K = .455 \quad C_2 = 0$

$F_{SU} = 28{,}000$ psi $\quad f_s = \frac{795}{.035} = 22{,}700$ psi.

$M.S. = \frac{28.0}{22.7} - 1 = \underline{+.23}$

STA. 365 to 403 $\quad q = 986$ LBS/IN. $\quad t = .040$ INS.

$d/t = \frac{5.0}{.040} = 125 \quad h/t = 26/.04 = 650 \quad K = .45 \quad C_2 = 0$

$F_{SU} = 28{,}000$ psi $\quad f_s = \frac{986}{.04} = 24{,}600$ psi.

$M.S. = \frac{28.0}{24.6} - 1 = \underline{+.14}$

STA. 403 to 475 $\quad q = 760$ LBS/IN $\quad t = .032$ INS.

$d/t = \frac{5.0}{.032} = 156 \quad h/t = \frac{26}{.032} = 810 \quad K = .50 \quad C_2 = 0$

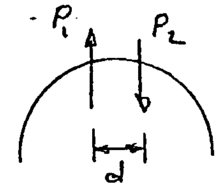
$F_{SU} = 27{,}800$ psi. $\quad f_s = 760/.032 = 23{,}700$ psi.

$M.S. = \frac{27.8}{23.7} - 1 = \underline{+.17}$

STA. 475 AFT $\quad q = 93$ LBS/IN. $\quad t = .025$ INS. (MIN.)

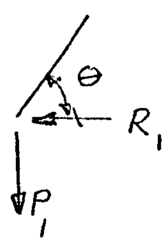
SUPPORT STRUCTURE, VERTICAL TAIL

FRAME COUPLE LOAD



FRAME	d	$\bar{Y} = d/2$	CAP AREA	$A\bar{Y}$	M/I	P_1 & P_2 (LBS.)
FWD. (@ F. SPAR)	9.4	4.7	.24	1.13	10,720	12,100
CENTER (@ C. SPAR)	11.0	5.5	.36	1.98	10,720	21,200
AFT (@ A. SPAR)	6.8	3.4	.24	.816	10,720	8,750

LOAD IN UPPER SKIN PANEL

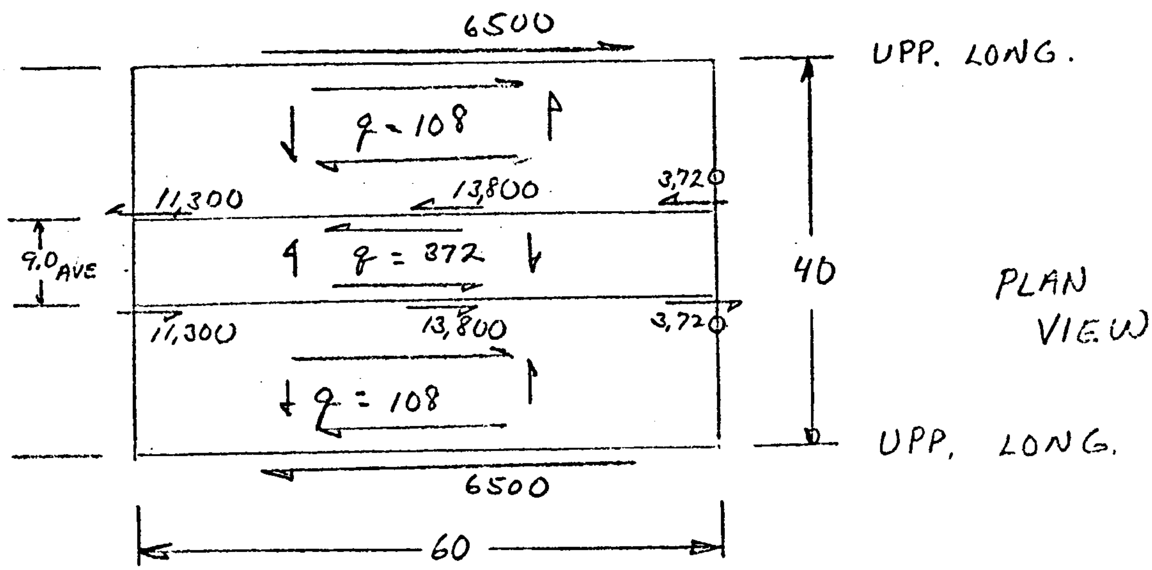


$$R_1 = P_1 \times \tan\theta$$

FRAME	θ	$\tan\theta$	R_1 (LBS)
FWD.	47°	.931	11,300
CENTER	57°	.650	13,800
AFT	67°	.425	3,720

SUPPORT STRUCTURE, VERTICAL TAIL

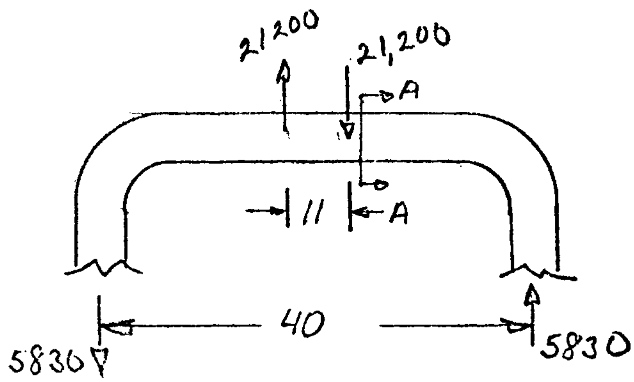
UPPER SKIN PANEL



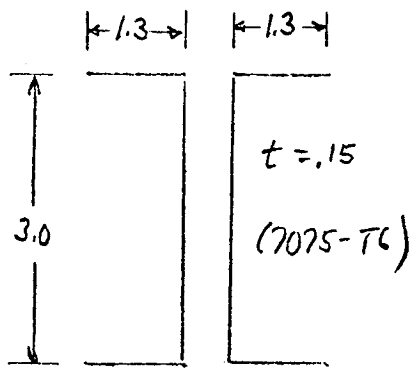
SKIN t = .030 INS. MIN. REQD.

SUPPORT STRUCTURE, VERTICAL TAIL

FRAME @ CENTER SPAR



$M_{A-A} = (5{,}830)(14.5) = 84{,}500$ IN. LBS.



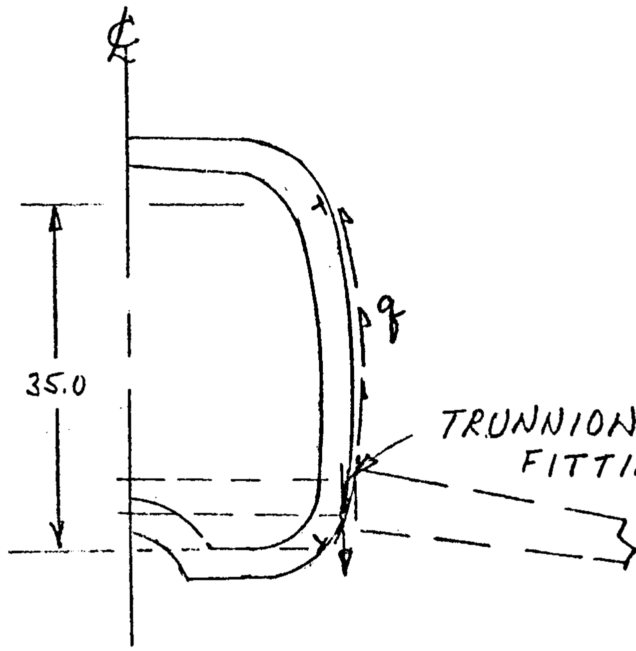
$I = 2.07 \text{ IN}^4$

$f_{bc} = (84{,}500)(1.5)/2.07 = 61{,}200 \text{ psi}$

$b/t = \frac{1.225}{.15} = 8.1 \qquad F_{MAX} = 62{,}000 \text{ psi}$

$M.S. = \frac{62.0}{61.2} - 1 = \underline{+.01}$

SUPPORT FRAMES, HORIZ. TAIL

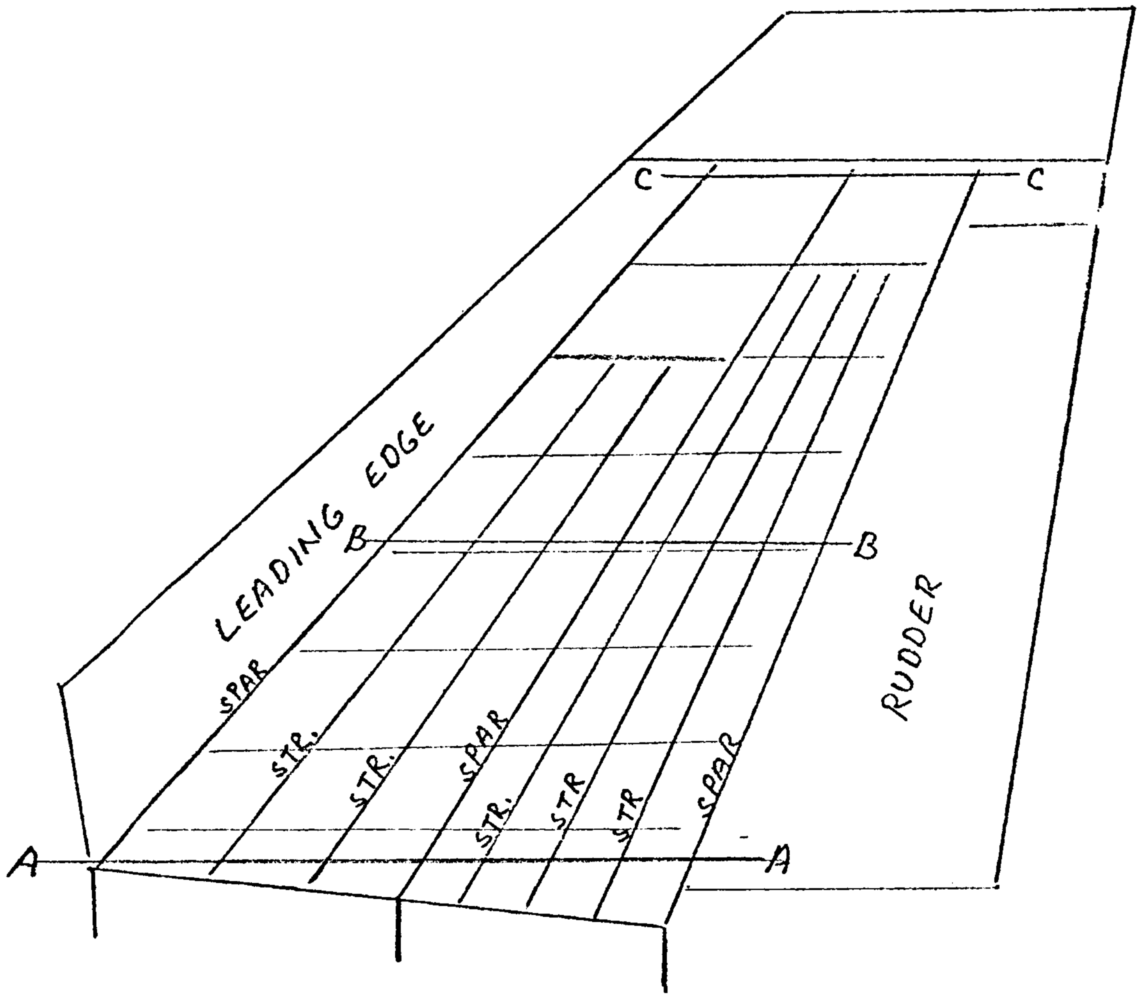


LOAD @ TRUNNION FITTING = 1930 LBS. ULT.

LOAD / FRAME = $1930/2$ = 965 LBS.

$q = 965/35 = 27.6$ LBS/IN.

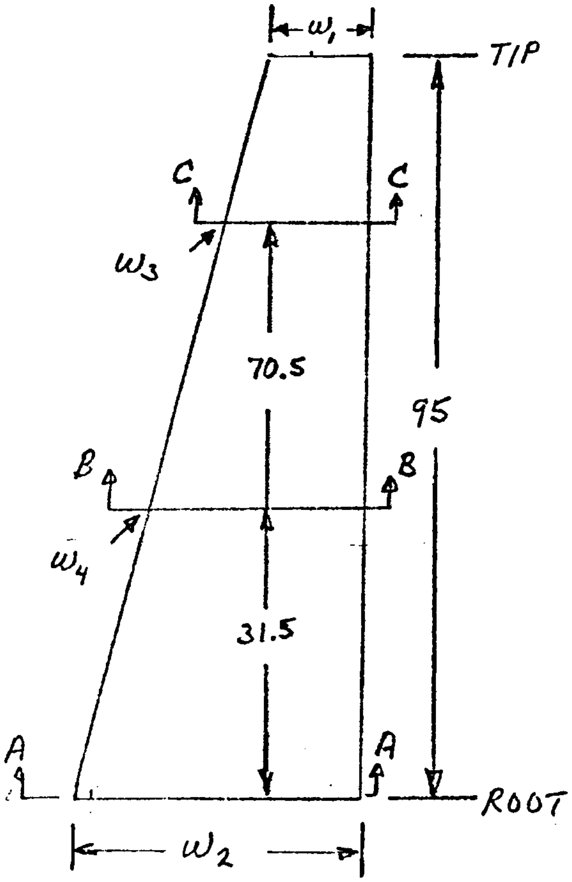
O.K. BY INSPECTION



VERTICAL TAIL

LOAD = (6,230)(1.5) = 9,350 LBS. ULT.
@ 25% MAC

SPANWISE LOADING



EQ. ① $95w_1 + \frac{1}{2}(w_2 - w_1)(95) = 9350$

EQ ② $(95w_1)(47.5) + \left[\frac{1}{2}(w_2 - w_1)(95)\right](31.7) = (9350)(40)$

SOLVING EQUATIONS

w_1 = 52 LBS/IN.

w_2 = 145 LBS/IN.

w_3 = 24 LBS/IN

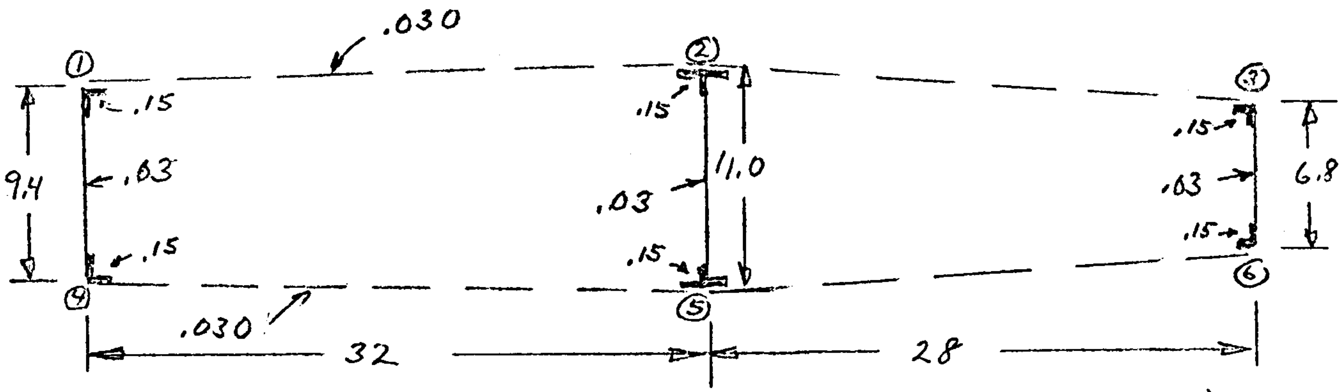
w_4 = 63 LBS/IN.

ULT. LOADS @ E.A. ·(.47c)

SECTION	V_Y (LBS)	M_X (IN. LBS)	M_Z (IN. LBS)
C-C	1,570	18,100	15,700
B-B	5,300	147,000	85,000
A-A	9,350	374,000	234,000

VERTICAL TAIL

SECTION A-A (EFFECTIVE BENDING SECTION)



(7075-T6)

ITEM	A	Y	AY	AY²
1	.24	4.5	1.08	4.86
2	.36	5.3	1.91	10.10
3	.24	3.2	.77	2.46
4	.24	-4.5	-1.08	4.86
5	.36	-5.3	-1.91	10.10
6	.24	-3.2	-.77	2.46
Σ	1.68		—	34.84

$f_{bc} = (374{,}000)(5.5)/34.84$

$= 59{,}000$ psi

$F_{MAX} = 64{,}000$ psi

$M.S. = \frac{64}{59} - 1 = +\underline{.08}$

$EI = (10.3 \times 10^6)(34.84) = 360 \times 10^6$ LB. IN.²

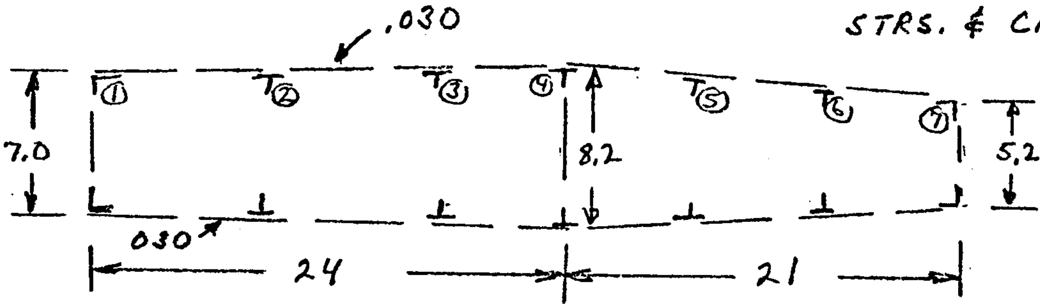
$$J = \frac{4A^2}{\sum \frac{\Delta s}{t}} = \frac{(4)(\overline{572}^2)}{136/.03} = 288 \text{ IN.}^4$$

$GJ = (3.9 \times 10^6)(288) = 1{,}120 \times 10^6$ LB. IN.²

$$q_T = \frac{T}{2A} = \frac{234{,}000}{(2)(572)} = 204 \text{ LBS/IN.}$$ N.C.

SECTION B-B (EFFECTIVE BENDING SECTION)

STRS. & CAPS $t = .05$



(7075 - T6)

ITEM	A	Y	AY	AY²
1	.094	3.4	.320	1.09
2	.094	3.6	.338	1.22
3	.094	3.8	.358	1.36
4	.094	4.0	.376	1.50
5	.094	3.5	.328	1.15
6	.094	3.0	.282	.84
7	.094	2.5	.234	.59
Σ	.658			7.75

$I = (2)(7.75) = 15.50 \text{ IN}^4$

$f_{bc} = (147{,}000)(4.1)/15.5$

$= 39{,}000 \text{ psi}$

$F_{CR} = 42{,}000 \text{ psi}$

$M.S. = \frac{42}{39} - 1 = \underline{\underline{+.08}}$

$EI = (10.3 \times 10^6)(15.50) = 160 \times 10^6 \text{ LB IN}^2$

$$J = \frac{4A^2}{\Sigma \frac{\Delta S}{t}} = \frac{(4)(323^2)}{102/.03} = 123 \text{ IN}^4$$

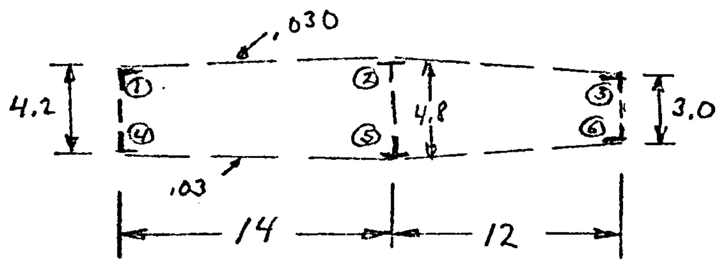
$GJ = (3.9 \times 10^6)(123) = 480 \times 10^6 \text{ LB. IN}^2$

$$q_T = \frac{T}{2A} = \frac{85{,}000}{(2)(323)} = 131 \text{ LBS/IN.}$$

N.C.

SECTION C-C (EFFECTIVE BENDING SECTION)

CAPS $t = .040$



(7075 - T6)

ITEM	A	Y	AY	AY^2
1	.090	2.0	.180	.360
2	.090	2.3	.207	.476
3	.090	1.4	.126	.176
4	.090	-2.0	-.180	.360
5	.090	-2.3	-.207	.476
6	.090	-1.4	-.126	.176
Σ	.540			2024

$$f_{bc} = (18,100)(2.4)/2.024$$

$$= 21,400 \text{ psi}$$

$$F_{MAX} = 25,000 \text{ psi}$$
$$(b/t = 17.5)$$

$$M.S. = \frac{25.0}{21.4} - 1 = \underline{+.17}$$

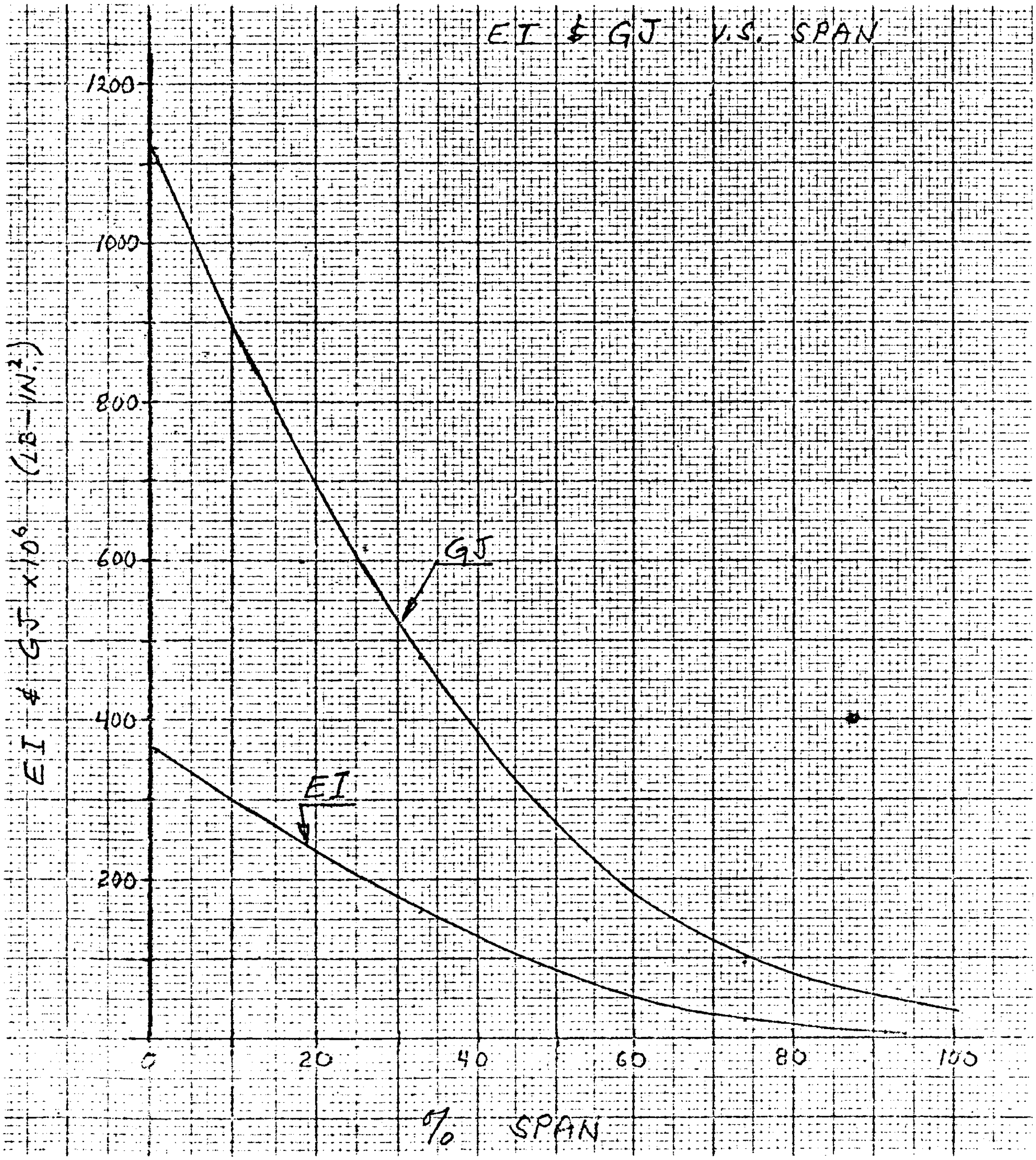
$$EI = (10.3 \times 10^6)(2.024) = 20.9 \times 10^6 \text{ LB. IN}^2$$

$$J = \frac{4A^2}{\Sigma \Delta s/t} = \frac{(4)(110^2)}{59/.03} \doteq 24.6 \text{ IN}^4$$

$$GJ = (3.9 \times 10^6)(24.6) = 96 \times 10^6 \text{ LB. IN}^2$$

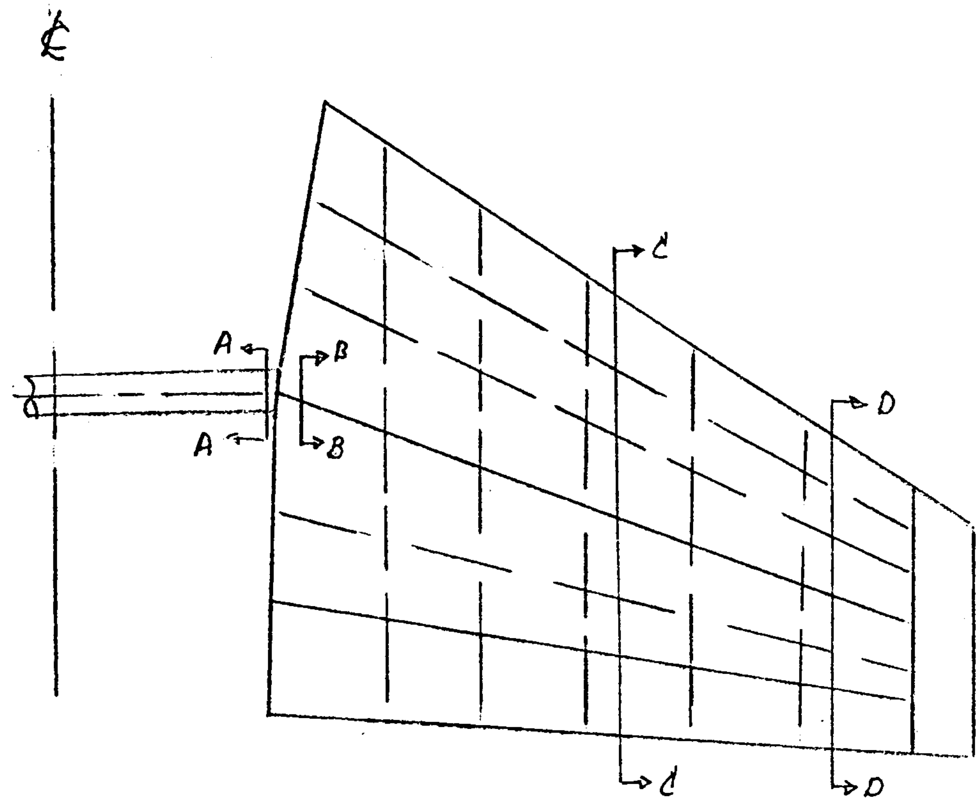
$$q_T = \frac{T}{2A} = \frac{15,700}{(2)(110)} = 71.5 \text{ LBS/IN}$$

N.C.



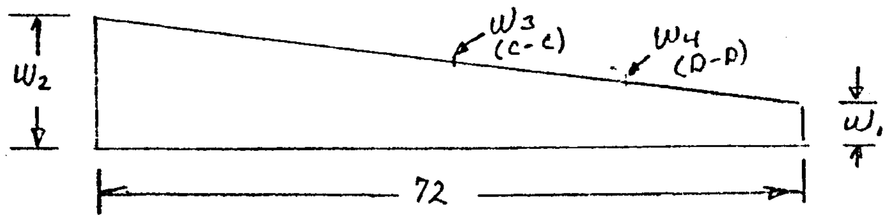
VERTICAL TAIL

PLAN VIEW



LOAD $= (1{,}285)(1.5) = 1{,}930$ LBS.. ULT. @ 25% M.A.C.

SPANWISE LOADING



EQ. ① $72\,W_1 + \frac{1}{2}(W_2 - W_1)(72) = 1930$

EQ ② $(72\,W_1)(36) + \left[\frac{1}{2}(W_2 - W_1)(72)\right] 24 = 1930 \times 30$

SOLVING EQUATIONS.

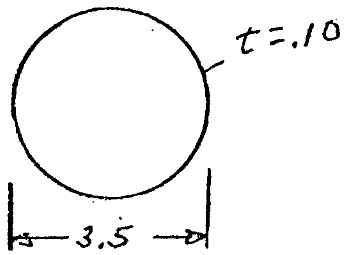
$W_1 = 13.6$ LBS/IN. $W_2 = 40.0$ LBS/IN.

$W_3 = 26.8$ LBS/IN $W_4 = 18.7$ LB/IN.

HORIZONTAL TAIL

SECTION A-A

$$M = (1930)(30) = 58{,}000 \text{ IN. LBS.}$$



(7075-T6)
(EXTR)

$$I = \frac{\pi}{4}(r_0^4 - r_1^4)$$

$$= 1.54 \text{ IN}^4$$

$$f_b = (58{,}000)(1.75)/1.54 = 66{,}000 \text{ psi}$$

$$F_{BU} = 75{,}000 + (.25)(65{,}000) = 91{,}000 \text{ psi}$$

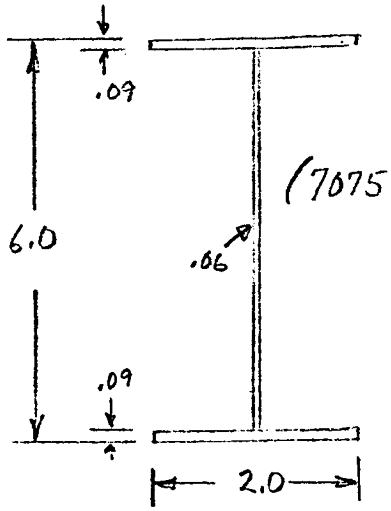
$$M.S. = \frac{91}{66} - 1 = \underline{\underline{+.38}}$$

$$EI = (10.3 \times 10^6)(1.54) = 15.9 \times 10^6 \text{ LB. IN}^2$$

$$GJ = (3.9 \times 10^6)(1.54 \times 2) = 12.0 \times 10^6 \text{ LB. IN}^2$$

HORIZONTAL TAIL

SECTION B-B



$$I = 2\left[(.18)(2.95^2)\right] + \tfrac{1}{12}(.06)(5.8^3)$$
$$= 4.11 \text{ IN}^4.$$

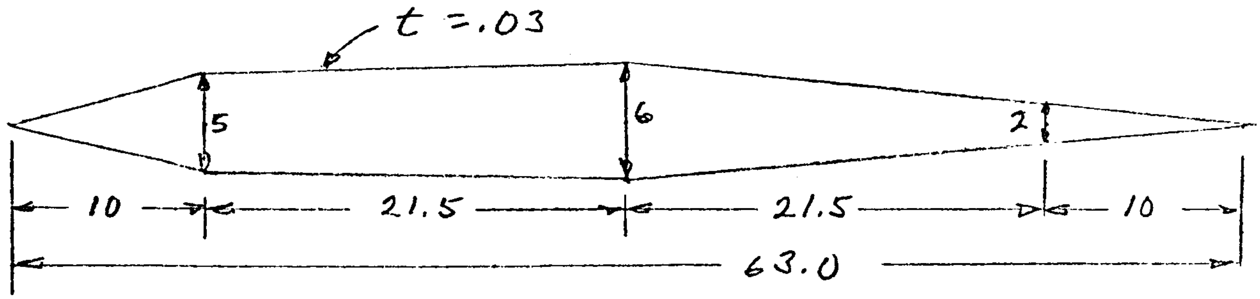
$$F_{bc} = (58{,}000)(3.0)/4.11 = 42{,}500 \text{ psi}$$

$$b/t = \frac{.97}{.09} = 10.8 \qquad F_{MAX} = 47{,}000 \text{ psi}$$

$$M.S. = \frac{47.0}{42.5} - 1 = \underline{+.11}$$

$$EI = (10.3 \times 10^6)(4.11) = 42.4 \times 10^6 \text{ LB. IN}^2$$

TORQUE SECTION @ B-B



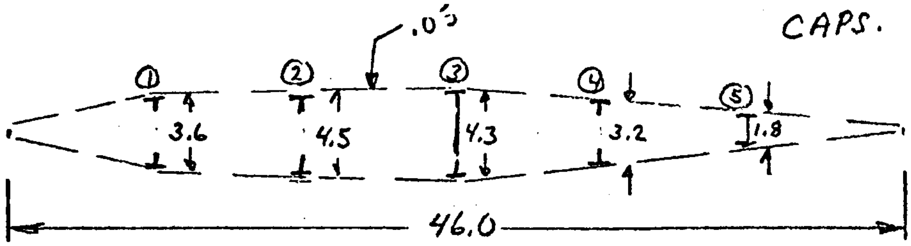
$$AREA = 239 \text{ IN}^2$$

$$J = \frac{4A^2}{\Sigma \Delta S/t} = \frac{(4)(239^2)}{130/.03} = 52.8 \text{ IN}^4$$

$$GJ = (3.9 \times 10^6)(52.8) = 206 \times 10^6 \text{ LB. IN}^2$$

SECTION C-C (EFFECTIVE BENDING SECTION)

CAPS. = .04



(7075-T6)

ITEM	A	Y	AY	AY^2
1	.06	1.70	.102	.173
2	.06	2.10	.126	.264
3	.06	2.00	.120	.240
4	.06	1.50	.090	.135
5	.06	.80	.048	.038
Σ	.30			.850

$I = (2)(.850) = 1.70 \text{ IN}^4$

$f_{b_c} = (11650)(2.15)/1.70$

$= 14{,}600$ psi

$b/t = \frac{.75}{.04} = 18.7$ $F_{MAX} = 21{,}000$ psi

$M.S. = \frac{21.0}{14.6} - 1 = \underline{\underline{+.44}}$

$V_Z = \frac{1}{2}(13.6 + 26.8)(36) = 730$ LBS.

$M_Y = (730)(12) = 8750$ IN. LBS.

$M_X = (13.6)(36)(18) + \frac{1}{2}(13.2)(36)(12) = 11{,}650$ IN. LBS

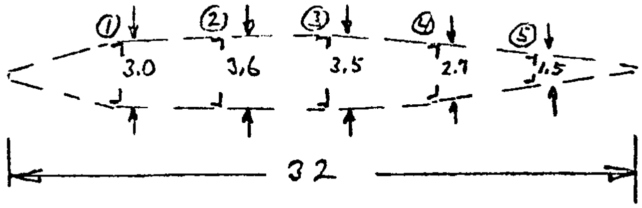
$EI = (10.3 \times 10^6)(1.7) = 17.5 \times 10^6$ LB. IN^2.

$J = \frac{(4)(\overline{130}^2)}{\Sigma\, 95/.03} = 21.4 \text{ IN}^4$

$GJ = (3.9 \times 10^6)(21.4) = 83.5$ LB. IN^2.

HORIZONTAL TAIL

SECTION D-D



$t_{SKIN} = .030$ } MIN.
$t_{CAPS} = .040$

(7075-T6)

ITEM	A	Y	AY	AY^2
1	.06	1.4	.084	.118
2	.06	1.7	.102	.173
3	.06	1.65	.099	.163
4	.06	1.25	.075	.094
5	.06	.65	.039	.025
Σ	.30			.573

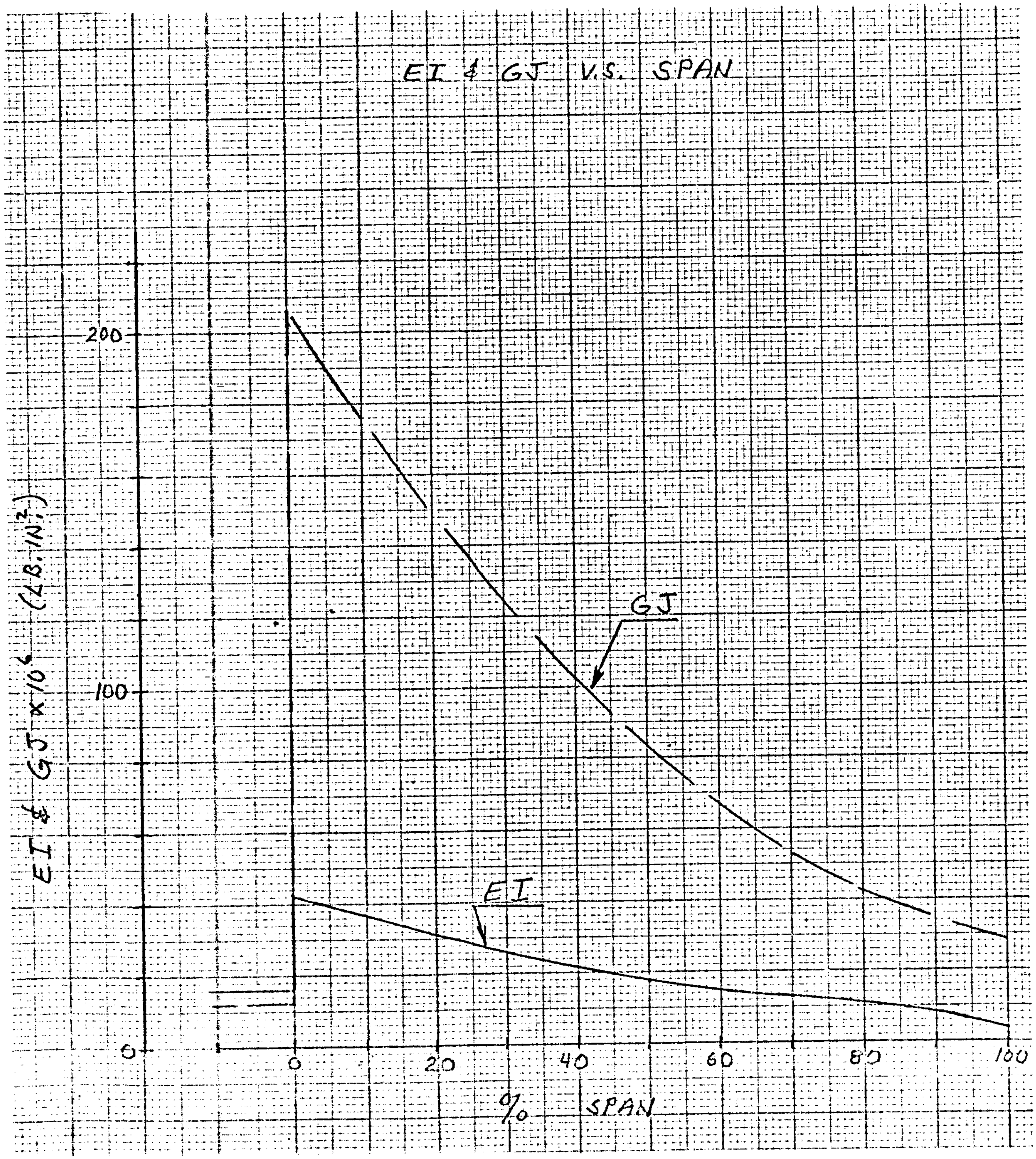
(SECTION NOT CRITICAL BENDING OR SHEAR)

$I = (2)(.573) = 1.146 \ IN^4$

$EI = (10.3 \times 10^6)(1.146) = 11.8 \times 10^6 \ LB. IN^2$

$J = \frac{(4)(\overline{76}^2)}{\Sigma\, 65/.03} = 10.7 \ IN^4$

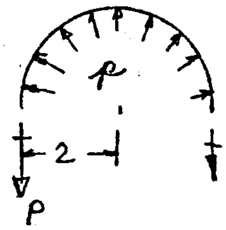
$GJ = (3.9 \times 10^6)(10.7) = 41.7 \ LB \cdot IN^2$



HORIZONTAL TAIL

WING DUCT DIA. = 4.0 INS.

p = 75 psi. ULT T = 500°F



$P = (75)(2) = 150$ LBS.

$t = .020$ MIN.
321 STAINLESS STL.

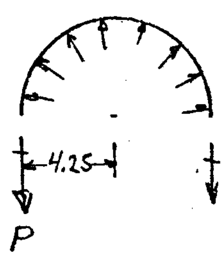
$f_t = 150/.02 = 7500$ psi

$F_{TU} = (75{,}000)(.76) = 57{,}000$ psi

$M.S. = \frac{57.0}{7.5} - 1 = +6.6$

FUSELAGE DUCT DIA = 8.5 INS.

p = 75 psi. ULT. T = 500°F



$P = (75)(4.25) = 318$ LBS.

$t = .02$

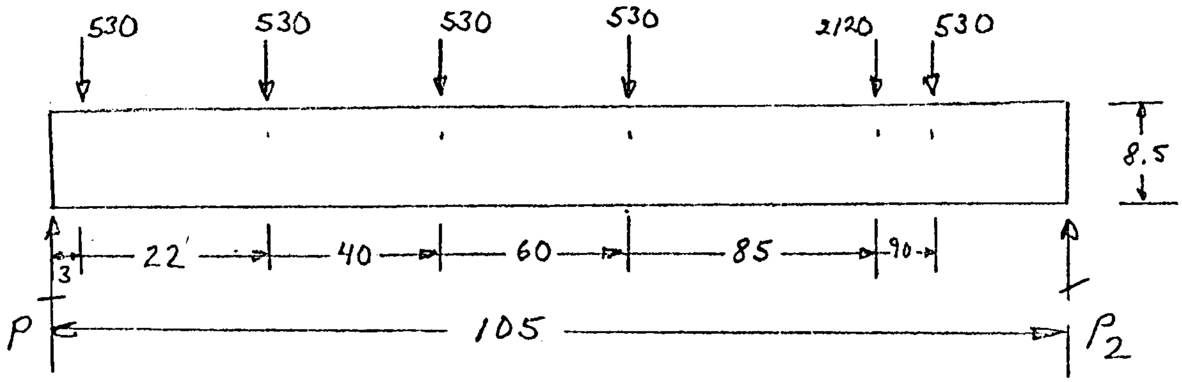
$f_t = 318/.02 = 15{,}900$ psi.

$M.S. = \frac{57}{15.9} - 1 = \underline{+2.58}$

FUSELAGE (CENTER BLEED MANIFOLD)

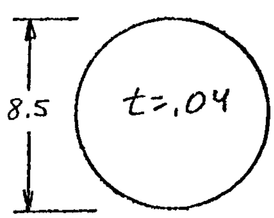
LOAD – 3.0 IN. DIA. PIPE = $(7.07)(75) = 530$ LBS

LOAD – 6.0 IN. DIA. PIPE = $(28.2)(75) = 2120$ LBS.



$P_1 = 1{,}970$ LBS. $\quad P_2 = 2{,}800$ LBS.

$M_{MAX} = (1970)(60) - (530)(115) = 57{,}000$ IN. LBS.



321 STAINLESS STEEL

$I = \frac{\pi}{4}\left(4.25^4 - 4.21^4\right) = 9.50 \text{ IN}^4$

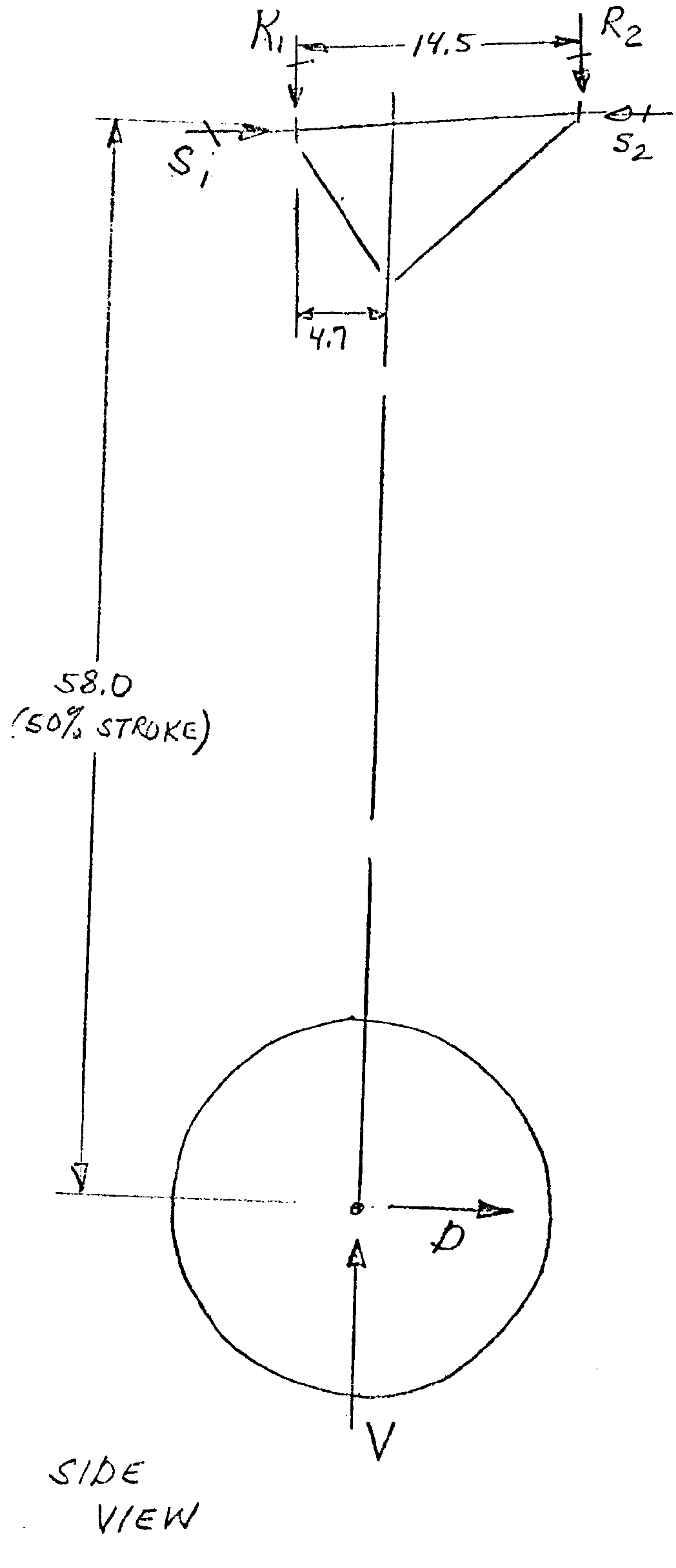
$f_b = (57{,}000)(4.25)/9.5 = 25{,}500$ psi

$F_{tu} = (75{,}000)(.76) = 57{,}000$ psi

$F_{cCR} = .3\,E\left(\frac{t}{R}\right) = (.3)(29\times10^6)\left(\frac{.04}{4.25}\right) = 82{,}000$ psi

$(82{,}000)(.76) = 62{,}500$ psi

$M.S. = \frac{57.0}{25.5} - 1 = \underline{+1.23}$



COND. VERTICAL LDG.
@ 15 FPS SINK SPEED

V = 22,700 LBS
D = -11,350 LBS.

$R_2 = \frac{1}{14.5}\left[22{,}700 \times 4.7 - 11{,}350 \times 58\right]$

R_2 = -38,100 LBS.

R_1 = 60,800 LBS.

S_1 = 11,350 LBS.

V = 22,700 LBS.
D = +11,350 LBS.

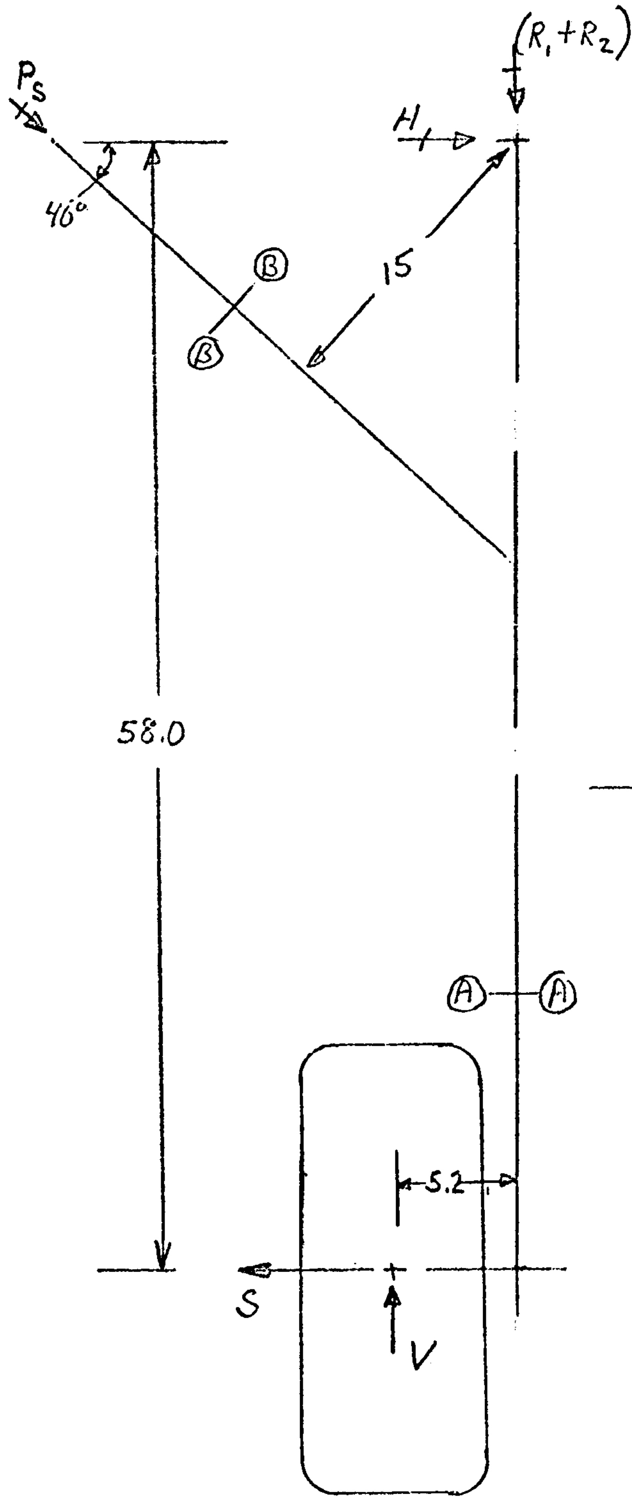
$R_2 = \frac{1}{14.5}\left[22{,}700 \times 4.7 + 11{,}350 \times 58\right]$

R_2 = 53,000 LBS.

R_1 = -30,300 LBS.

S_2 = 11,350 LBS

MAIN LANDING GEAR



$V = 22{,}700$ LBS.
$S = 11{,}350$ LBS.

$P_s = \frac{1}{15}[22{,}700 \times 5.2 + 11{,}350 \times 58]$

$P_s = 51{,}800$ LBS.

$H = (-51{,}800 \cos 40 + 11350)$

$H = -28{,}350$ LBS.

$(R_1 + R_2) = 22{,}700 - (51{,}800)(\sin 40)$

$(R_1 + R_2) = -10{,}600$ LBS.

$V = 22{,}700$ LBS.
$S = -11{,}350$ LBS.

$P_s = \frac{1}{15}[(22{,}700)(5.2) - 11{,}350 \times 58]$

$= -36{,}200$ LBS.

$H = (36\,200 \cos 40 - 11{,}350)$

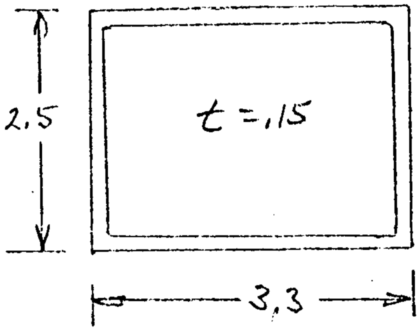
$= 16{,}450$ LBS

$(R_1 + R_2) = 22{,}700 + (36{,}200)(\sin 40)$

$(R_1 + R_2) = 46{,}000$ LBS

MAIN LANDING GEAR

SECTION A-A



$$I = \frac{(3.3)(2.5^3)}{12} - \frac{(3.0)(2.2^3)}{12} = 1.63 \text{ IN.}^4$$

$$M = (11{,}350)(10) + (22{,}700)(6.5)$$
$$= 260{,}500 \text{ IN. LBS.}$$

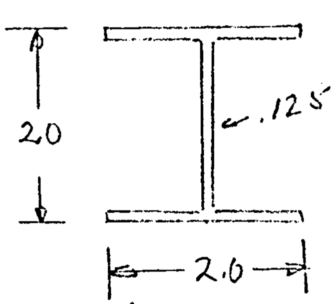
STL. HT. 260-280,000 psi.

$$f_b = \frac{(260{,}500)(1.25)}{1.63} = 199{,}000 \text{ psi}$$

$$F_{cy} = 242{,}000 \text{ psi}$$

$$M.S. = \frac{242}{199} - 1 = \underline{\underline{+.21}}$$

SECTION B-B



(STL. HT. 260-280)

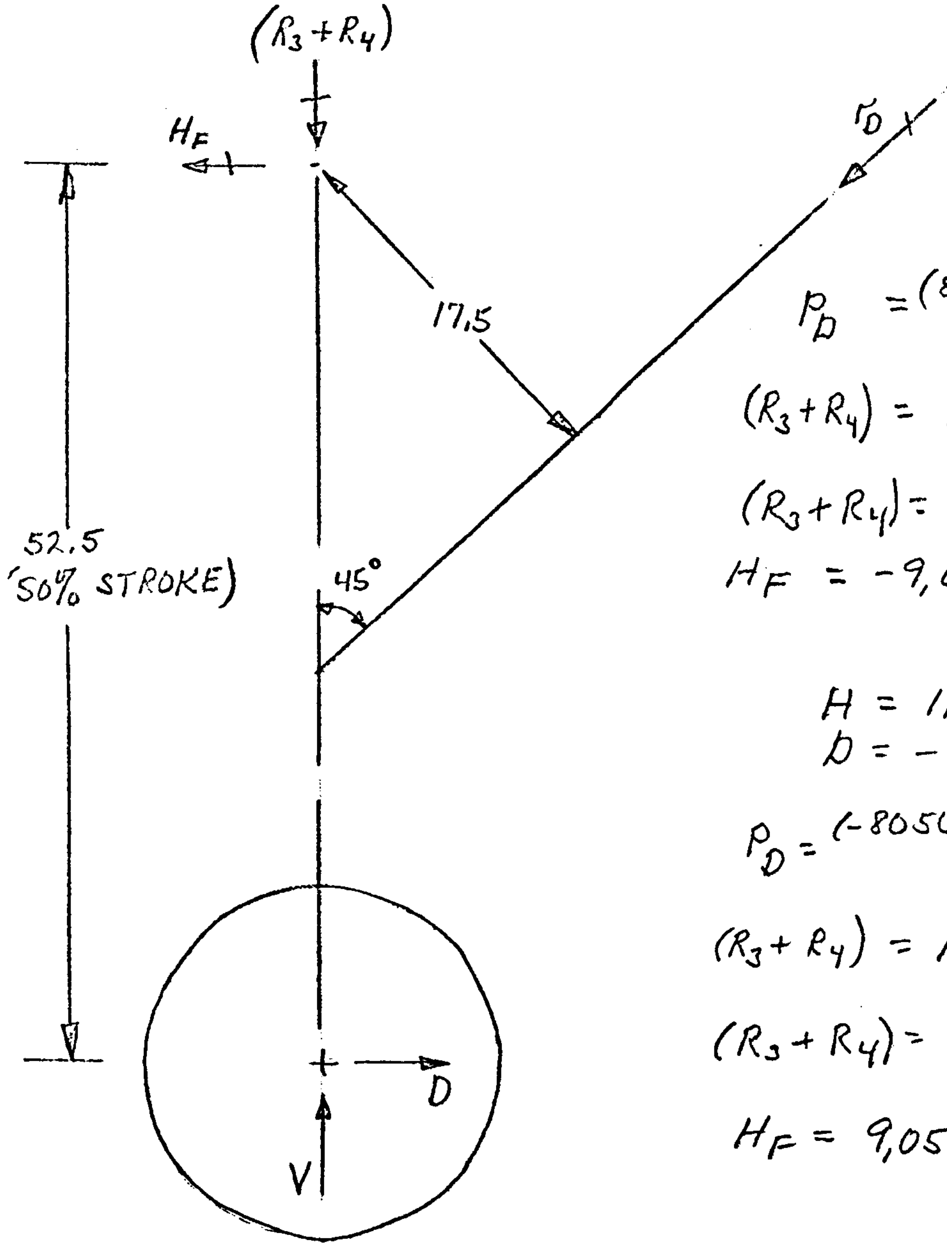
$$I = .495 \text{ IN}^4$$
$$A = .719 \text{ IN}^2$$
$$\rho = .830$$

$$f_c = 51{,}800/.719 = 72{,}000 \text{ psi}$$

$$L/\rho = \frac{36.5}{.83} = 44.0$$

$$F_c = 89{,}000 \text{ psi}$$

$$M.S. = \frac{89}{72} - 1 = \underline{\underline{+.23}}$$



$V = 11{,}500$ LBS
$D = 8{,}050$ LBS.

$P_D = (8050)(52.5)/17.5 = 24{,}150$ LBS.

$(R_3+R_4) = 11{,}500 - (24{,}150)(\cos 45)$

$(R_3+R_4) = -5{,}600$ LBS.

$H_F = -9{,}050$ LBS.

$H = 11{,}500$ LBS.
$D = -8{,}050$ LBS

$P_D = (-8050)(52.5)/17.5 = -24{,}150$ LBS

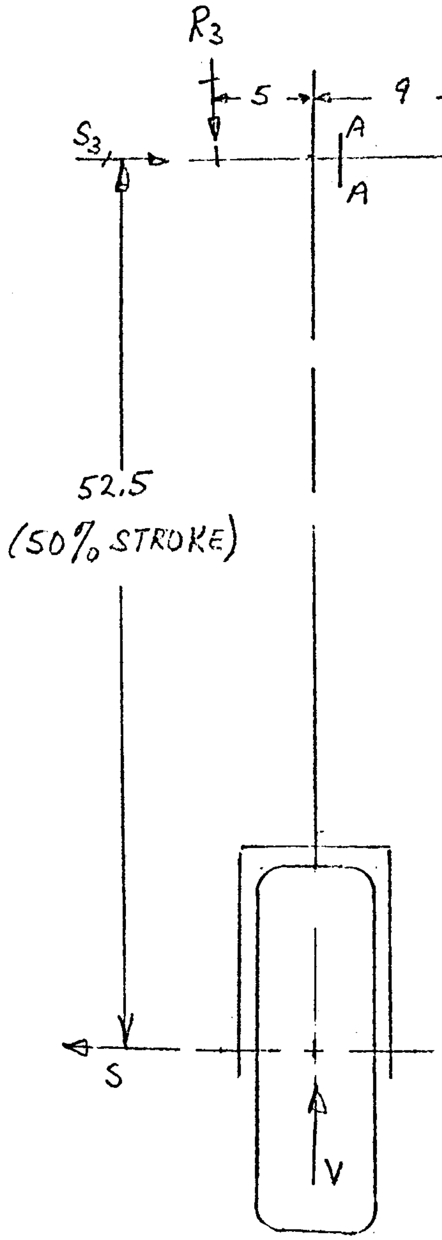
$(R_3+R_4) = 11{,}500 - (-24{,}150)(\cos 45)$

$(R_3+R_4) = 28{,}600$ LBS.

$H_F = 9{,}050$ LBS.

SIDE VIEW

NOSE LANDING GEAR



$V = 11,500$ LBS.
$S = 8,050$ LBS.

$R_3 = \frac{1}{14}[11,500 \times 9 + 8050 \times 52.5]$

$= 37,600$ LBS.

$R_4 = 11,500 - 37,600 = -26,100$ LBS

$S_3 = 8050$ LBS.

$V = 11,500$ LBS.
$S = -8,050$ LBS.

$R_3 = \frac{1}{14}[11,500 \times 9 - 8050 \times 52.5]$

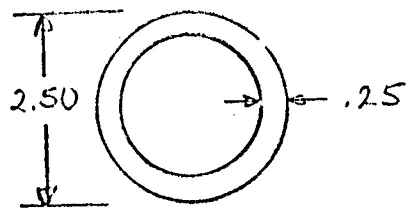
$R_3 = -22,800$ LBS.

$R_4 = 11500 + 22,800 = 34,300$ LBS.

$S_4 = 8,050$ LBS.

NOSE LANDING GEAR

SECTION A-A



$I = 1.132 \text{ IN}^4$

$M = (34,300)(6) = 206,000$ IN.LBS

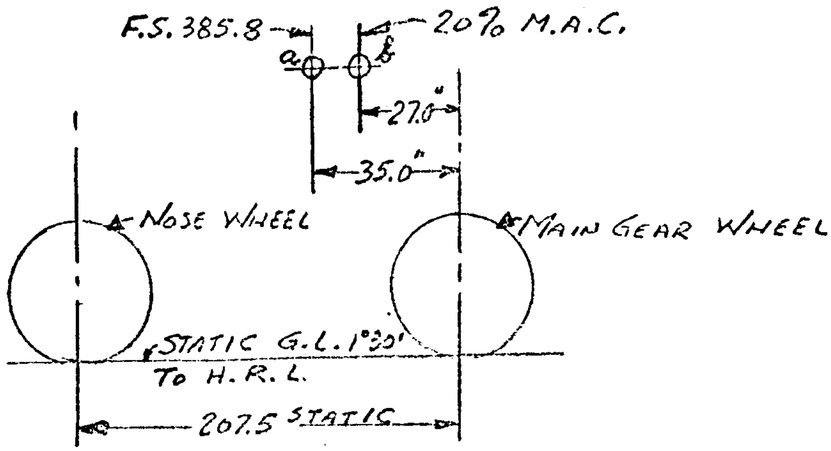
STL. HT 260-280,000 psi

$F_{cy} = 242,000$ psi

$f_b = \frac{(206,000)(1.25)}{1.132} = 228,000$ psi.

$M.S. = \frac{242}{228} - 1 = \underline{+.06}$

NOSE LANDING GEAR



LANDING GEAR STATIC LOADS

NOSE GEAR TIRE LOAD

a) $P_{NS} = \dfrac{18000 \times 35}{207.5} = \underline{\underline{3040}}$ #

b) $P_{NS} = \dfrac{18,000 \times 27}{207.5} = 2340$ #

MAIN GEAR TIRE LOAD

a) $P_{MS} = \dfrac{18,000 - 3040}{2} = 7480$ #

b) $P_{MS} = \dfrac{18,000 - 2340}{2} = \underline{\underline{7830}}$ #

DATA (N309)

1) G.W. = 18,000 #

2) MAIN GEAR TIRE SIZE = 22 × 8.5 - 11

3) M.G. IMPACT TIRE DEFLECTION ALLOWABLE = $\left(\dfrac{22-11}{2} - .812\right) \times .65 = 3.04''$

MAIN GEAR LANDING LOAD FACTOR @ 15 FT/SEC. SINK RATE, 2 POINT LANDING = 2.5g LIMIT

1.) $K.E. = \dfrac{9000 \times \overline{15}^2}{64.4} = 31,450$ #/SIDE

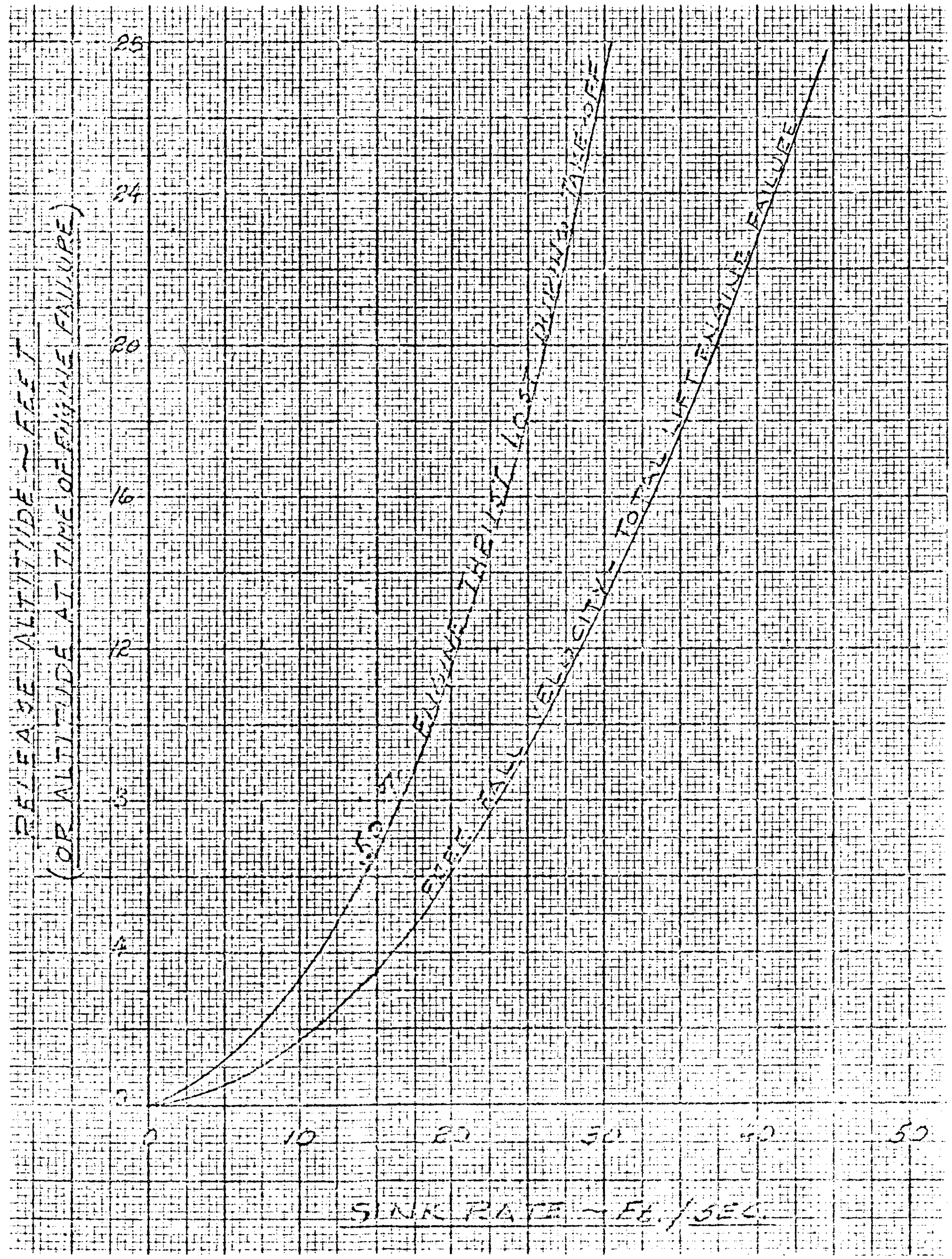
2) $E_S + E_T = \dfrac{31,450}{9,000 \times 2} = 1.40'$ (TOTAL STROKE REQ'D)

$E_S = 1.4 - E_T = 1.4 - \left(\dfrac{3.0}{12} \times .50\right) = 1.275'$

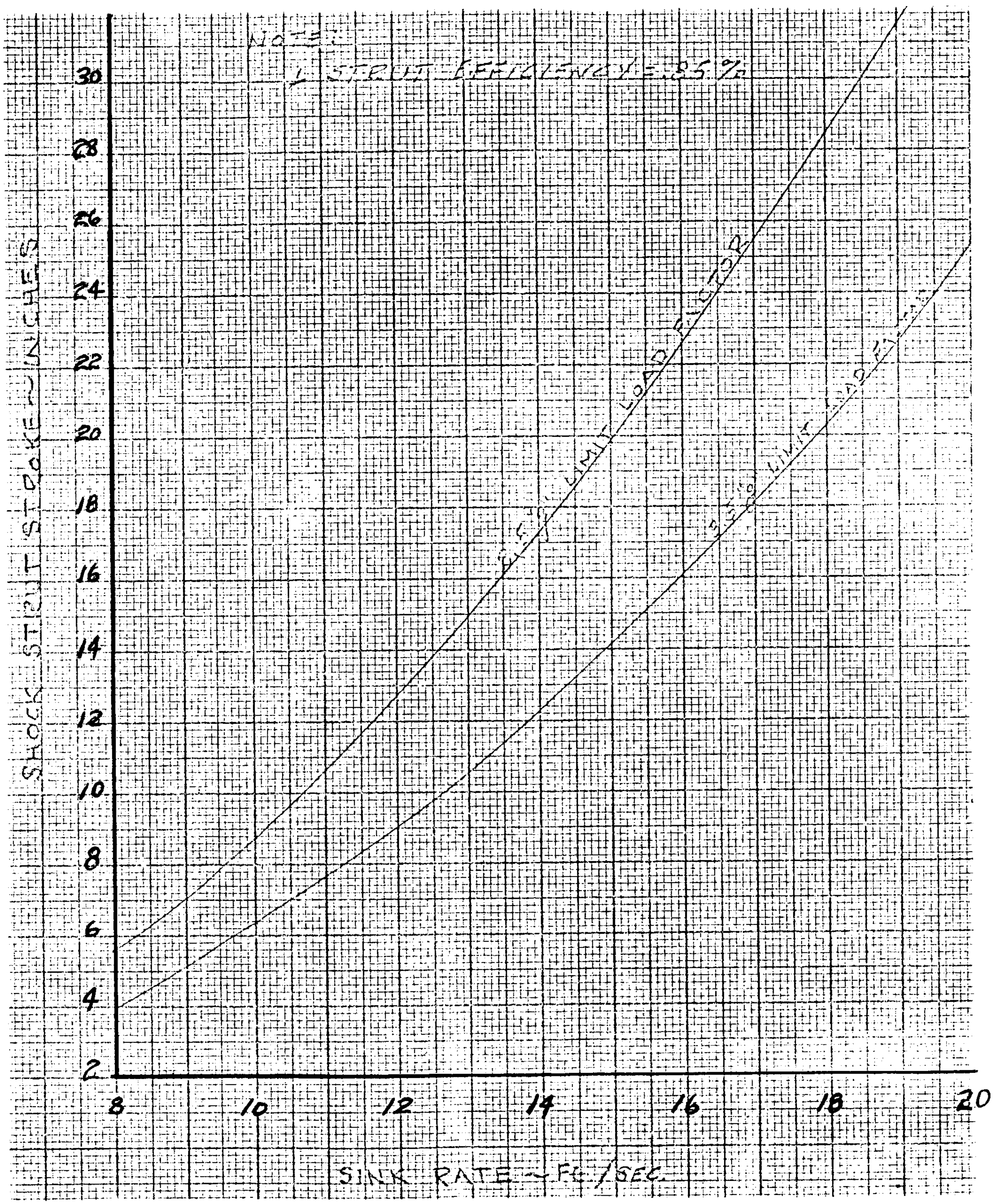
or

S_S (ASSUMING SHOCK STRUT EFF = 80%) $= \dfrac{1.275 \times 12}{.80} = 19.15''$

LANDING GEAR



SINK RATE VS ALTITUDE AS RELATED WITH ENGINE THRUST FAILURE DURING VTOL FLIGHT



AIRPLANE LANDING SINK RATE VS SHOCK STRUT STROKE

APPENDIX 5-II

NASA V/STOL PHASE III

MASS PROPERTIES DATA

NORTHROP CORPORATION
NORAIR DIVISION

NASA V/STOL

Phase III

Mass Properties Data

PREPARED BY

C. W. Glose

E. E. Robinson

APPROVED BY

REVISIONS

CHANGE	DATE	ENGINEER	PAGES AFFECTED OR REMARKS

FORM 20-701 (R.10-63)

Foreword

This report is one of a series of reports submitted by Northrop Corporation - Norair Division per contract No. NAS1-6777 for the V/STOL Jet Operations Research Airplane Design Study.

Issued in general accordance with the requirements of MIL-W-25140 (ASG) for an estimated weight report. Justification of the weight estimate, accuracy and contingency analyses are included.

Definitions

New Airplane

N-309 - Drawing No. AD-4486A
VTOL Weight 18,000 lbs.
2 J85-19 Lift Cruise Engines
7 J85-19 Lift Engines (Space for 8)

Modified Airplane

T-39A - Drawing No. AD-4448A
VTOL Weight 20,200 lbs.
2 J85-19 Lift Cruise Engines
7 J85-19 Lift Engines (Space for 10)

FORM 20-7A
(R.11-63)

ENGINEER	NORTHROP CORPORATION NORAIR DIVISION	PAGE
CHECKER		REPORT NO.
DATE		MODEL

Section I

N-309
Drawing No. AD-4486A
New Airplane

(See Page 1-3 of this report or Page 5-II-92 of this section)

FORM 20-7A
(R.11-63)

ENGINEER	NORTHROP CORPORATION NORAIR DIVISION	PAGE
CHECKER		REPORT NO.
DATE	Phase III Mass Properties Data	MODEL

Table of Contents

ENGINEER

CHECKER

DATE

NORTHROP CORPORATION
NORAIR DIVISION

Weight, Center of Gravity & Moment of Inertia Summary

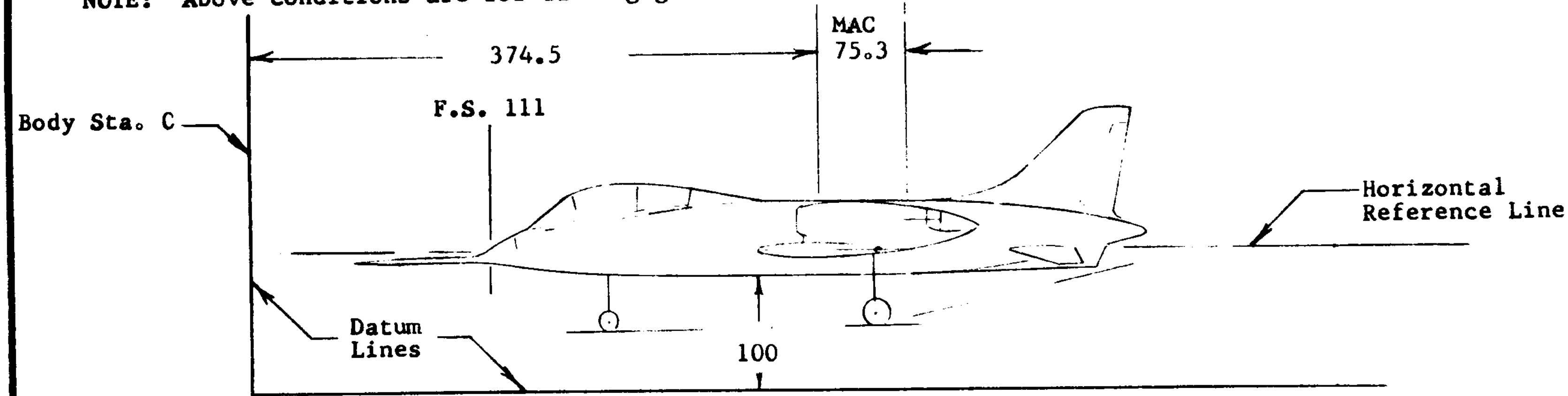
PAGE 1

REPORT NO.

MODEL N-309

Condition	Weight, Pounds	Center of Gravity			Moment of Inertia, Slug-Ft^2			Product of Inertia, Izz Sl.Ft^2	Principal Axis Inclination
		Horizontal Fus.Sta.	Horizontal % MAC	Vertical Water Line	Roll Ixx	Pitch Iyy	Yaw Izz		
Composite Mode:									
Gross Wt. Less Fuel	13,993	385.3	14.3	110.0	9,052	37,593	43,373	243	$0°24.5'$
Usable Fuel	4,007	396.7	--	114.9	--	--	--	-	--
VTOL Gross Weight	18,000	387.8	17.7	111.1	9,326	45,662	51,668	70	$0°3.0'$
Direct Lift Mode:									
Gross Wt. Less Fuel	13,896	382.6	10.8	109.9	9,117	36,110	41,955	170	$0°18.0'$
Usable Fuel	2,404	397.0	--	104.3	--	--	--	-	--
VTOL Gross Weight	16,300	384.7	13.5	109.1	9,227	41,122	47,042	18	$0°1.0'$

NOTE: Above conditions are for landing gear down and a crew of two.



REFERENCE PLANE DIAGRAM

	Direct Lift	Lift/Cruise	AD-4486A Composite J-85s
Structure			(4,568)
Wing			930
Tail			309
Body (incl. air induction & exhaust doors)			2,537
Alighting Gear			792
Propulsion & Nacelle	(3,261)	(2,159)	(5,420)
Engine Installation	2,709	784	
Swivel Nozzle/Diverter Valve	231	150	
Engine Section, Nacelles & Pylons	189	491	
Air Induction	35	36	
Exhaust & Cooling	14	138	
Lubrication System	7	2	
Engine Controls	50	28	
Starting System	26	15	
Constant Speed Drives	--	140	
Fuel System	--	375	
Power Systems			(1,561)
Surface Controls			508
Hydraulics			190
Electrical			335
Hover Controls (ducting & valves)			528
Equipment Groups			(869)
Instruments			171
Electronics			220
Furnishings & Equipment			410
Air Conditioning & Anti-Icing			68
Contingency			(267)
Weight Empty			12,685
Operating Weight Empty Items			(508)
Crew (2)			400
Unusable & Trapped Fuel			45
Oil			63
Operating Weight Empty			13,193
Usable Fuel			4,007
Payload			800
Maximum VTOL/Gross Weight			18,000

N-309
AD-4486A
Composite Mode
New Airplane

GROUP WEIGHT STATEMENT

ESTIMATED - ~~XXXXXXXXXXXXXXXX~~

(Cross out those not applicable)

CONTRACT NO.

AIRPLANE, GOVERNMENT NO.

AIRPLANE, CONTRACTOR NO.

MANUFACTURED BY

		MAIN	AUXILIARY
ENGINE	MANUFACTURED BY	General Electric	General Electric
	MODEL	J85-19	J85-19
	NO.	2	7*
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

*Provisions for 8

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

No.	Item				Detail	Group Total
1	WING GROUP					930
2	CENTER SECTION - BASIC STRUCTURE				158	
3	INTERMEDIATE PANEL - BASIC STRUCTURE					
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS 0 LBS.)				548	
5						
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM 0 LBS.)				82	
7	AILERONS (INCL. BALANCE WEIGHT 0 LBS.)				30	
8	FLAPS - TRAILING EDGE				51	
9	- LEADING EDGE				61	
10	SLATS					
11	SPOILERS					
12	SPEED BRAKES					
13						
14						
15	TAIL GROUP					309
16	STABILIZER - BASIC STRUCTURE - All Moveable				144	
17	FINS - BASIC STRUCTURE (INCL. DORSAL LBS.)				97	
18	SECONDARY STRUCTURE (STAB. & FINS)					
19	ELEVATOR (INCL. BALANCE WEIGHT LBS.)				33	
20	RUDDERS (INCL. BALANCE WEIGHT 0 LBS.)					
21	Dorsal Fairing				7	
22	Ventral Fin/Skid				28	
23	BODY GROUP					2,537
24	FUSELAGE OR HULL - BASIC STRUCTURE				1,751	
25	BOOMS - BASIC STRUCTURE					
26	SECONDARY STRUCTURE - FUSELAGE OR HULL				375	
27	- BOOMS					
28	- SPEEDBRAKES					
29	- DOORS, ~~[illegible]~~				411	
30						
31	ALIGHTING GEAR GROUP - LAND (TYPE: Tricycle)					792
32-33	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS		
34	Main	170	330	100	600	
35	Nose	26	159	7	192	
36						
37						
38						
39						
40	ALIGHTING GEAR GROUP - WATER					
41	LOCATION	FLOATS	STRUTS	CONTROLS		
42						
43						
44						
45						
46	SURFACE CONTROLS GROUP					508
47	COCKPIT CONTROLS				68	
48	~~[illegible]~~ Stability Augmentor				118	
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS All LBS.)				322	
50						
51	ENGINE SECTION OR NACELLE GROUP					680
52	INBOARD					
53	CENTER - Lift Engines				189	
54	OUTBOARD - Lift/Cruise Engines				146	
55	DOORS, PANELS & MISC. (Incl. in nacelles)					
56	Nacelles (Incl. 13 lbs. Access Doors)				345	
57	TOTAL (TO BE BROUGHT FORWARD)					5,756

AN-9103-D
NAME ____________
DATE ____________

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

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MODEL N-309
REPORT ____________

1	PROPULSION GROUP					4,740
2			AUXILIARY -Lift		MAIN -Cruise	
3	ENGINE INSTALLATION		2,709		784	
4	AFTERBURNERS (IF FURN. SEPARATELY)					
5	ACCESSORY GEAR BOXES & DRIVES - CSD				140	
6	SUPERCHARGERS (FOR TURBO TYPES)					
7	AIR INDUCTION SYSTEM		35		36	
8	EXHAUST SYSTEM		14		138	
9	X[redacted] Swivel Nozzle/Diverters		231		150	
10	LUBRICATING SYSTEM		7		2	
11	TANKS (Incl. in Eng. Wt.)					
12	COOLING INSTALLATION					
13	DUCTS, PLUMBING, ETC.	7		2		
14	FUEL SYSTEM				375	
15	TANKS - PROTECTED					
16	- UNPROTECTED			90		
17	PLUMBING, ETC.			285		
18	WATER INJECTION SYSTEM					
19	ENGINE CONTROLS		50		28	
20	STARTING SYSTEM		26		15	
21	PROPELLER INSTALLATION					
22						
23	Propulsion Sub-Totals		(3072)		(1668)	
24	AUXILIARY POWER PLANT GROUP					
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP					171
26	HYDRAULIC & PNEUMATIC GROUP					190
27	Hover Control System (Ducting & Valves)					528
28						
29	ELECTRICAL GROUP					335
30						
31						
32	ELECTRONICS GROUP					220
33	EQUIPMENT				174	
34	INSTALLATION				46	
35						
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)					
37	FURNISHINGS & EQUIPMENT GROUP					410
38	ACCOMMODATIONS FOR PERSONNEL				305	
39	MISCELLANEOUS EQUIPMENT				44	
40	FURNISHINGS				26	
41	EMERGENCY EQUIPMENT				35	
42						
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP					68
44	AIR CONDITIONING & Equipment Cooling				61	
45	ANTI-ICING - Cabin Defog				7	
46						
47	PHOTOGRAPHIC GROUP					
48	AUXILIARY GEAR GROUP					
49	HANDLING GEAR					
50	ARRESTING GEAR					
51	CATAPULTING GEAR					
52	ATO GEAR					
53						
54						
55	[redacted] Contingency					267
56	TOTAL FROM PG. 2					5,756
57	WEIGHT EMPTY					12,685

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NAME
DATE

GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

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REPORT

No.	Load Condition				Composite Mode VTOL	Composite Mode Operating Wt. Empty	Direct Lift Mode VTOL	Direct Lift Mode Operating Wt. Empty
1	LOAD CONDITION				Composite Mode		Direct Lift Mode	
2					VTOL	Operating Wt. Empty	VTOL	Operating Wt. Empty
3	CREW (NO. 2)				400	400	400	400
4	PASSENGERS (NO.)							
5	FUEL	Type	Gals.					
6	UNUSABLE	JP-4	6.9		45	45	45	45
7	INTERNAL-Flight	JP-4	616.5		4,007			
8	-Flight	JP-4	369.8				2,404	
9								
10	EXTERNAL							
11								
12	BOMB BAY							
13								
14	OIL (9 pounds per engine)				63	63	70	70
15	TRAPPED							
16	ENGINE							
17								
18	FUEL TANKS (LOCATION)							
19	WATER INJECTION FLUID (GALS)							
20								
21	BAGGAGE							
22	CARGO							
23								
24	ARMAMENT							
25	GUNS (Location)	Fix. or Flex.	Qty.	Cal.				
26								
27								
28								
29								
30								
31								
32	AMMUNITION							
33								
34								
35								
36								
37								
38								
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)							
*40	BOMB OR TORPEDO RACKS							
41								
42	Variable Stability System				500		500	
43	NASA Research Equipment				300		---	
44								
45								
46	EQUIPMENT							
47	PYROTECHNICS							
48	PHOTOGRAPHIC							
49								
*50	OXYGEN							
51								
52	MISCELLANEOUS							
53								
54								
55	USEFUL LOAD				5,315	508	3,419	515
56	WEIGHT EMPTY				12,685	12,685	12,881	12,881
57	GROSS WEIGHT				18,000	13,193	16,300	13,396

*If not specified as weight empty.

AN-9103-D
NAME
DATE

GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

PAGE 7
MODEL N-309
REPORT

1 LENGTH - OVERALL (FT.) 55.58 | HEIGHT - OVERALL - STATIC (FT.) 15.33

No.	Item	Main Floats	Aux. Floats	Booms	Fuse or Hull	Nacelles Inboard	Nacelles Center	Nacelles Outboard
2								
3	LENGTH - MAX. (FT.)				46.25	121		
4	DEPTH - MAX. (FT.)				5.250	40		
5	WIDTH - MAX. (FT.)				5.667	28		
6	WETTED AREA (SQ. FT.)				723	150		
*7	FLOAT OR HULL DISPL. - MAX (LBS.)							

8 FUSELAGE VOLUME (CU. FT.) | PRESSURIZED | TOTAL

No.	Item	Wing	H. Tail	V. Tail
9				
10	GROSS AREA (SQ. FT.)	210	66.6	48.0
11	WEIGHT/GROSS AREA (LBS./SQ. FT.)	4.42	1.95	3.0
12	SPAN (FT.)	35.5	15.7	7.6
13	FOLDED SPAN (FT.)			
14				
15	SWEEPBACK - AT 25% CHORD LINE (DEGREES)	20°	27°	40°
16	- AT % CHORD LINE (DEGREES)			
**17	THEORETICAL ROOT CHORD - LENGTH (INCHES)	101.5	78.5	117.0
18	- MAX. THICKNESS (INCHES)	13.195	7.85	11.70
**19	CHORD AT PLANFORM BREAK - LENGTH (INCHES)			
20	- MAX. THICKNESS (INCHES)			
***21	THEORETICAL TIP CHORD - LENGTH (INCHES)	40.6	24.0	35.0
22	- MAX. THICKNESS (INCHES)	4.872	2.40	3.50

23 DORSAL AREA, INCLUDED IN XXXXXXX(HULL) (V. TAIL) AREA (SQ. FT.) 3.5

24 TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.) 15.0

No.	Item						
25	AREAS (SQ. FT.)	Flaps	L.E. 23.2	T.E. 23.2			
26		Lateral Controls	Slots	Spoilers	Ailerons 8.8 (aft of hinge line)		
27		Speed Brakes	Wing	Fuse. or Hull			
28							
29							

No.	ALIGHTING GEAR (LOCATION)	Nose	Main		
30	ALIGHTING GEAR	Nose	Main		
31	LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)	62.5	68.0		
32	OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)	20.0	19.25		
33	FLOAT OR SKI STRUT LENGTH (INCHES)				

34 ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)

35 HYDRAULIC SYSTEM CAPACITY (GALS.)

No.	FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
36	FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
37	Fuel - Internal	Wing				
38		Fuse. or Hull			6	707.7
39	- External					
40	- Bomb Bay					
41						
42	Oil					
43						
44						

No.	STRUCTURAL DATA - CONDITION	Fuel in Wings (Lbs.)	Stress Gross Weight	Ult. L.F.
45	STRUCTURAL DATA - CONDITION	Fuel in Wings (Lbs.)	Stress Gross Weight	Ult. L.F.
46	FLIGHT	0	18,000	5.625
47	LANDING			
48				
49	MAX. GROSS WEIGHT WITH ZERO WING FUEL			
50	CATAPULTING			
51	MIN. FLYING WEIGHT			
52	LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)			

53 WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)

54 STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)

55 PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)

56

57 AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.) 7,837 lbs.

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ at ℄ airplane.
***Parallel to ℄ airplane.
****Total usable capacity.

N-309
AD-4486A
Composite Mode
New Airplane

DETAIL WEIGHT STATEMENT

ESTIMATED - ~~XXXXXXXXXXXXXXX~~

(Cross out those not applicable)

CONTRACT NO. ______________

AIRPLANE, GOVERNMENT NO. ______________

AIRPLANE, CONTRACTOR NO. ______________

MANUFACTURED BY ______________

		MAIN	AUXILIARY
ENGINE	MANUFACTURED BY	General Electric	General Electric
	MODEL	J85-19	J85-19
	NO.	2	7*
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

*Provisions for 8

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WING GROUP
BASIC STRUCTURE

		Center Section	Interm. Panel	Outer Panel		
1						
2						
3	CODE NO.					
4	UPPER - SPAR CAP - FRONT					
5	- INTERMEDIATE					
6	- REAR					
7	- AUXILIARY					
8	- INTERSPAR COVERING	29		58		
9	SPANWISE STIFFENERS	19		66		
10	- JOINTS, SPLICES & FASTENERS	27		21		
11						
12						
13						
14	LOWER - SPAR CAP - FRONT					
15	- INTERMEDIATE					
16	- REAR					
17	- AUXILIARY					
18	- INTERSPAR COVERING	17		46		
19	SPANWISE STIFFENERS	19		66		
20	- JOINTS, SPLICES & FASTENERS	27		21		
21						
22						
23						
24	SPAR WEB & STIFFENERS - FRONT	5		15		
25	- INTERMEDIATE					
26	- REAR	5		15		
27	- AUXILIARY					
28	- JOINTS, SPLICES & FASTENERS					
29						
30	Wing Trunnion Ribs			38		
31	Landing Gear Trunnion Ribs			28		
32	INTERSPAR - RIBS	10		27		
33	- BULKHEADS					
34	- CHORDWISE STIFFENERS					
35						
36	LEADING EDGE - COVERING			23		
37	- STIFFENERS			19		
38	- RIBS			5		
39	- AUXILIARY SPARS			5		
40	- JOINTS, SPLICES & FASTENERS			5		
41						
42						
43	TRAILING EDGE - COVERING			36		
44	- STIFFENERS			30		
45	- RIBS			8		
46	- AUXILIARY SPARS			8		
47	- JOINTS, SPLICES & FASTENERS			8		
48						
49						
50	TIPS					
51						
52	FIREWALL (STRUCTURAL)					
53						
54						
55						
56	TOTALS - BASIC STRUCTURE	158		548		
57	TOTAL (TO BE BROUGHT FORWARD)					706

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WING GROUP
SECONDARY STRUCTURE
(DOORS, PANELS & MISCELLANEOUS)

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		* Location	** Type Power	Area Sq. Ft.	Structure	Operating Mechanism: Mechanism & Controls	Power Trans.	Actuator	Lock Mech.	Emerg.
1										
2										
3										
4	CODE NO.									
5	WING FOLD									
6										
7										
8	DOORS & FRAMES-LANDING	Strut	Strut	11	32					
9	- Landing	Wing	H	10	29	21				
10										
11	- BOMB									
12										
13										
14	- GUN									
15										
16	- AMMUNITION									
17										
18	- ROCKET									
19										
20	- LIFE RAFT									
21										
22	- ESCAPE									
23										
24	- ACCESS									
25										
26										
27										
28	PANELS-(NON STRUCTURAL)									
29										
30										
31										
32										
33										
34										
35										
36										
37										
38										
39										
40										
41										
42										
43										
44										
45										
46										
47										
48										
49	WALKWAYS, STEPS, GRIPS									
50										
51	FAIRING & FILLETS									
52										
53	EXTERIOR FINISH									
54										
55										
56	TOTALS				61	21				
57	TOTAL - SECONDARY STRUCTURE (TO BE BROUGHT FORWARD)									82

*Indicate location for major doors by C.S., O.P., I.P., etc.

**H-Hydraulic, E-Electrical, P-Pneumatic; power transmission from main distribution point to actuating unit.

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WING GROUP
CONTROL SURFACES

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	Ailerons Inb'd	Ailerons Outb'd	T.E. Flaps Inb'd	T.E. Flaps Outb'd	L.E. Flaps or Slats Inb'd	L.E. Flaps or Slats Outb'd	Spoilers	Speed Brakes	
1									
2									
3									
4 CODE NO.									
5 SPARS				3		4			
6									
7									
8									
9 RIBS		4		8		10			
10									
11									
12 COVERING & STIFFENERS		16		26		30			
13									
14									
15 T.E. STRIPS									
16									
17 FABRIC & DOPE									
18									
19									
20									
21 TABS									
22									
23									
24									
25 TORQUE TUBES									
26									
27									
28									
29 BALANCE WEIGHTS & SUPPORTS									
30									
31 AERO. SEAL									
32									
33									
34 CONTROL HORNS									
35									
36									
37 ACCESS DOORS (NON STRUCT.)									
38									
39 HINGES & PINS		5		6		8			
40 EXTERIOR FINISH									
41 TOTALS - SURFACE	( )	(25)	( )	(43)	( )	(52)	( )	( )	( )
42									
43 CONTROL SURFACE SUPPORTS									
44 HINGES		5		8		9			
45 BRACKETS									
46 TRACKS									
47 CARRIAGES									
48									
49									
50									
51									
52 TOTALS - SUPPORTS	( )	(5)	( )	(8)	( )	(9)	( )	( )	( )
53 TOTALS (LINES 41 & 52)		30		51		61			
54 TOTALS - CONTROL SURFACES		30		51		61			
55 TOTAL									142
56 TOTALS FROM PGS. 2 & 3									788
57 TOTAL - WING GROUP									930

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TAIL GROUP
BASIC STRUCTURE

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		Stabilizer		Fins			
1							
2		C.S.	O.P.	Center	Ventral	Dorsal	
3	CODE NO.						
4	UPPER - SPAR CAP - FRONT						
5	- INTERMEDIATE						
6	- REAR						
7	- AUXILIARY						
8	- INTERSPAR COVERING						
9	SPANWISE STIFFENERS						
10	- JOINTS, SPLICES & FASTENERS						
11	Spars - Front (incl. caps)			4			
12	- Center (incl. caps)	6		4			
13	- Rear (incl. caps)	3		3			
14	LOWER - SPAR CAP - FRONT						
15	- INTERMEDIATE						
16	- REAR						
17	- AUXILIARY						
18	Total - INTERSPAR COVERING	37		19			
19	Total SPANWISE STIFFENERS	5		10			
20	Total - JOINTS SPLICES & FASTENERS	8		8			
21							
22							
23							
24	SPAR WEB & STIFFENERS - FRONT						
25	- INTERMEDIATE						
26	- REAR						
27	- AUXILIARY						
28	- JOINTS, SPLICES & FASTENERS						
29							
30							
31							
32	INTERSPAR - RIBS	13		16			
33	- BULKHEADS						
34	- CHORDWISE STIFFENERS						
35							
36	LEADING EDGE - COVERING			9			
37	- STIFFENERS			2			
38	- RIBS			3			
39	- AUXILIARY SPARS						
40	- JOINTS, SPLICES & FASTENERS						
41							
42							
43	TRAILING EDGE - COVERING			3			
44	- STIFFENERS						
45	- RIBS						
46	- AUXILIARY SPARS						
47	- JOINTS, SPLICES & FASTENERS						
48	Trailing Edge Strip	7					
49							
50	TIPS	6		8			
51							
52	Torque Tube Installation	55					
53	Miscellaneous	4		8			
54							
55	TOTALS	144		97	28	7	
56	TOTALS - BASIC STRUCTURE	144		132			
57	TOTAL (TO BE BROUGHT FORWARD)						276

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TAIL GROUP
CONTROL SURFACES

			Rudders			
1						
2		Elevators	Center	Outer		
3	CODE NO.					
4	SPARS		7			
5						
6						
7						
8	RIBS		1			
9						
10						
11						
12	COVERING & STIFFENERS		8			
13						
14						
15	T. E. STRIPS		4			
16						
17	FABRIC & DOPE					
18						
19						
20						
21	TABS					
22						
23	Honeycomb Core & Bonding		9			
24						
25	TORQUE TUBES					
26						
27						
28						
29	BALANCE WEIGHTS & SUPPORTS					
30						
31	AERO. SEAL					
32						
33						
34	CONTROL HORNS		1			
35						
36						
37	ACCESS DOORS (NON STRUCTURAL)					
38						
39	HINGES & PINS					
40	EXTERIOR FINISH					
41						
42	TOTALS - SURFACE	( )	(30)	( )	( )	( )
43						
44	CONTROL SURFACE SUPPORTS					
45	HINGES		3			
46	BRACKETS					
47						
48						
49						
50						
51						
52	TOTALS - SUPPORTS	( )	(3)	( )	( )	( )
53	TOTALS (LINES 42 & 52)		33			
54	TOTALS - CONTROL SURFACES		33			
55	TOTAL					33
56	TOTALS FROM PGS. 5 & 6					276
57	TOTAL - TAIL GROUP					309

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BODY GROUP
BASIC STRUCTURE

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		Stations				Fuselage or Hull				Booms
1										
2	CODE NO.					111-271	271-507	507-aft		
3	SECTION					Fwd	Center	Aft		
*4	**BULKHEADS & FRAMES**									
5	Bulkhead	112				12				
6	Bulkhead	271				31				
7	Bulkhead		310				25			
8	Bulkhead		340				25			
9	Bulkhead		369				25			
10	Bulkhead		398				25			
11	Bulkhead		429				25			
12	Bulkhead		465				30			
13	Bulkhead		507				31			
14	Frame			542				19		
15	Frame			578				29		
16	Frame			605				17		
17										
18										
19										
20										
21										
22										
23										
24	**MINOR FRAMES**					50	215	43		
25	**JOINTS, SPLICES & FASTENERS**					13	62	13		
26	**OVERTURN STRUCTURE**									
27										
28	**COVERING - UPPER BETWEEN LONGERONS**									
29	**- SIDE BETWEEN LONGERONS**									
30	**- LOWER BETWEEN LONGERONS**									
31										
32	**COVERING LONGITUDINAL STIFFENERS - UPPER BETW. LONG.**						53	21		
33	**- SIDE BETW. LONG.**					26	52	21		
34	**- LOWER BETW. LONG.**					30	49	15		
35										
36										
37	**LONGERONS - UPPER**					14	30	20		
38	**- LOWER**					12	30	23		
39	Engine Mount Structure						150			
40										
41	**LONGITUDINAL PARTITIONS - (STRUCTURAL)**									
42										
43	**FLOORING & SUPPORTS - (BASIC STRUCTURE)**					38	60	10		
44										
45	Miscellaneous					54	275	58		
46										
47	**FIREWALL - (STRUCTURAL)**						20			
48										
49	**KEELSONS**									
50	**KEEL**									
51										
52	**CHINE & SPRAY STRIPS**									
53	**STEP ASSEMBLY**									
54	**STAIRWAY - (STRUCTURAL)**									
55	**TOTALS**					280	1182	289		
56	**TOTALS - BASIC STRUCTURE**						1751			
57	**TOTAL (TO BE BROUGHT FORWARD)**									1751

*List all main & watertight bulkheads & frames individually. Minor frames may be combined.

BODY GROUP
SECONDARY STRUCTURE

	Fuselage or Hull				Booms	Speed Brakes
1						
2 CODE NO.						
3 SECTION	Fwd	Center	Aft			
4						
5 ENCLOSURES (EXCLUDING TURRET ENCLOSURES)						
6 CANOPY	170					
7 CANOPY-OPERATING MECHANISM	80					
8 -RAILS						
* 9 -CYLINDERS, PLUMBING, FLUID						
10						
11 GUNNER - TAIL						
12						
13 BOMBARDIER						
14 SIGHTING BLISTERS						
15						
16 WINDSHIELD (EXCLUDING BULLET PROTECTION)	45					
17						
18 WINDOWS & PORTS INCL. FRAMES	25					
19						
20 Flooring & Equipment Supports	20	10	25			
21						
22						
23						
24						
25						
26						
27						
28 FLOORING & SUPPORTS (SECONDARY STRUCTURE)						
29						
30						
31 STAIRWAYS & LADDERS (FIXED)						
32						
33						
34 STERNPOST & FITTINGS						
35 NOSE BUMPER (HULL)						
36 RUBBING STRIPS						
37						
38						
39						
40 TAIL CONE						
41						
42						
43 SPEED BRAKES - STRUCTURE						
44 - SUPPORTS						
45						
46						
47						
48						
49						
50						
51						
52						
53						
54						
55 TOTALS	340	10	25			
56 TOTALS - SECONDARY STRUCTURE	375					
57 TOTAL (TO BE BROUGHT FORWARD)						375

*From main distribution point to actuating unit.

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BODY GROUP
SECONDARY STRUCTURE
(DOORS, PANELS & MISCELLANEOUS)

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		* Location	** Type Power	Area Sq. Ft.	Structure	Mechanism & Controls	Operating Mechanism Power Trans.	Actuator	Lock Mech.	Emerg.
1										
2										
3										
4	CODE NO.									
5	DOORS & FRAMES									
6	- LANDING	Nose	H	7.5	15	20				
7	- Air Intake		H	38	76	49				
8	- Exhaust		H	35	102	30				
9										
10	- BOMB									
11										
12										
13	- GUN									
14										
15	- AMMUNITION									
16										
17	- ROCKET									
18										
19	- LIFE RAFT									
20										
21	- ESCAPE									
22										
23										
24	- WATER TIGHT									
25										
26	- COMPARTMENT									
27										
28	- ENTRANCE									
29		-								
30										
31	- ACCESS	Fwd. Fus.		19	57					
32		Cntr. Fus.		7	21					
33		Aft Fus.		12	36					
34										
35	- ENGINE									
36										
37	- CAMERA									
38										
39	PANELS - (NON STRUCTURAL)									
40										
41										
42										
43										
44										
45										
46										
47										
48										
49	WALKWAYS, STEPS, GRIPS				5					
50										
51	FAIRING & FILLETS									
52	EXTERIOR FINISH									
53										
54	TOTALS				312	99				
55	TOTAL - SECONDARY STRUCTURE (DOORS, PANELS, MISC.)									411
56	TOTALS FROM PGS. 8 & 9									2126
57	TOTAL - BODY GROUP									2537

*Indicate location for major doors by B - Booms, F or H for Fuselage or Hull.

**H - Hydraulic, E - Electrical, P - Pneumatic; power transmission from main distribution point to actuating unit.

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NAME ______________________

DATE ______________________

ALIGHTING GEAR GROUP

MODEL N-309

REPORT ______________________

Line	Item	Nose	Main		
1					
2	TYPE: Tricycle — *LOCATION	Nose	Main		
3	CODE NO.				
4	*LOCATION				
5	No. / Size / No. / Size / No. / Size				
6	WHEELS — 1 / 18 x 5.5 / 2 / 22 x 8.5	12	46		
7	TIRES — 1 / 18 x 5.5 / 2 / 22 x 8.5	14	49		
8	TUBES				
9	AIR				
10	BRAKES - Modified F-5A		75		
11	NO. & TYPE				
12	ENERGY CAP. **				
13	ANTI-SKID DEVICE				
14					
15	FLOATS - BULKHEADS				
16	- FRAMES				
17	- COVERING				
18	- COVERING STIFFENERS (LONGITUDINAL)				
19	- KEELSONS				
20	- KEEL				
21	- LONGITUDINAL PARTITIONS				
22	- CHINE & SPRAY STRIP				
23	- STEP ASSEMBLY				
24	- POST ASSEMBLY				
25	- NOSE BUMPER				
26	INSPECTION DOORS				
27	WALKWAYS				
28	EXTERIOR FINISH				
29	SKIDS OR BUMPERS				
30	SKIIS				
31					
32	TOTALS - RUNNING GEAR	(26)	(170)	( )	( )
33					
34	STRUTS - DRAG	26			
35	- SIDE		77		
36	- FLOAT				
37	PYLON				
38	SHOCK STRUT - STRUT (INCL. LBS. OIL)	127	245		
39	- FORK				
40	- AXLE				
41	- TORQUE ARMS				
42	- TRUNNIONS				
43	SHIMMY DAMPER OR SNUBBER	6			
44	Shrink Mechanism		8		
45	FITTINGS - MAIN ATTACH. - WING				
46	- TAIL				
47	- BODY				
48	- NACELLE				
49					
50	FAIRING				
51					
52					
53					
54					
55	PINS, BOLTS, NUTS, ETC.				
56	TOTALS - STRUCTURE	(159)	(330)	( )	( )
57	TOTALS (LINES 32 & 56) (TO BE BROUGHT FORWARD)	185	500		

*Descriptive location - Nose, Tail, Main, Outrigger, Bumper, etc.

**Ft. lbs./brake

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ALIGHTING GEAR GROUP
CONTROLS

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1 2 **LOCATION	Nose			Main							
3 4	Retract	Brake Oper.	Emerg. Ext.	Steering	Retract	Brake Oper.	Emerg. Ext.	Retract	Emerg. Ext.	Retract	Emerg. Ext.
5 CODE NO.											
6 MECHANICAL OPER. MECH.											
7 CONTROLS							17				
8 ACTUATORS											
9											
10											
11											
12 ELECTRICAL OPER. MECH.											
13 CONTROLS											
*14 WIRING, CONDUIT, ETC.					2						
15 OPERATING MOTORS											
16 MECHANISM											
17 Switches					7						
18											
19											
20 HYDRAULIC OPER. MECH.											
21 CONTROLS											
*22 PLUMBING & FLUID	7				13	21					
23 PUMPS											
24 RESERVOIRS											
25 ACCUMULATORS											
26 ACTUATORS					13	10					
27 MECHANISM											
28											
29											
30											
31 PNEUMATIC OPER. MECH.											
32 CONTROLS											
*33 PLUMBING											
34 PUMPS											
35 BOTTLES (AIR)											
36 ACTUATORS											
37 MECHANISM											
38											
39											
40											
41 LOCKING MECHANISM					17						
42 BRACES											
43 LINKS											
44 PARKING BRAKE CONTROL											
45 POSITION INDICATING MECH.											
46											
47											
48 SUPP'TS, GUIDES, ETC. - WING											
49 - TAIL											
50 - BODY											
51 - NACELLE											
52											
53 TOTALS	7				52	31	17				
54 TOTALS - CONTROLS	7			100							
55 TOTALS FROM PG. 11	185			500							
56 TOTALS	192			600							
57 TOTAL - ALIGHTING GEAR GROUP										792	

*From main distribution point to actuating unit.

**Descriptive location - Nose, Tail, Main, Outrigger, Bumper, etc.

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SURFACE CONTROLS GROUP
COCKPIT & AUTOPILOT

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		Cockpit Controls	Stability Augmentor XXXXXX
1			
2			
3	CODE NO.		
4	CONTROL COLUMNS		
5	CONTROL STICK OR COLUMN (PILOT)	14	
6	CONTROL STICK OR COLUMN (ASSIST. PILOT)	14	
7	CONNECTING MEMBERS FOR ABOVE (includes supports)	7	
8	SUPPORTS		
9			
10			
11			
12			
13	RUDDER PEDALS		
14	RUDDER PEDALS, INCLUDING BRAKE TREADLE (PILOT)	15	
15	RUDDER PEDALS, INCLUDING BRAKE TREADLE (ASSIST. PILOT)	15	
16	CONNECTING MEMBERS FOR ABOVE (includes supports)	3	
17	SUPPORTS		
18			
19			
20			
21			
22			
23	INTEGRAL PARKING LOCK		
24	CONTROL STICK		
25	RUDDER PEDALS		
26	SURFACES		
27			
28			
29			
30			
31			
32			
33	AUTOPILOT OR AUTO. FLIGHT CONTROL (TYPE: 3 Axis SAS)		
34	CONTROLLER - Cockpit (2 Required)		6
35	XXXXXXXXXX Electronic Package		16
36	SERVO AMPLIFIER		
37	SERVO MOTORS		
38	GYROS - Rate		7
39	Single Channel Servo Actuators (6 req.)		26
40	Solenoid shut-off valves (6 req.)		9
41	Trim mechanism		5
42			
43			
44			
45			
46	SUPPORTS & BRACKETS		
47			
*48	PLUMBING & FLUID		15
*49	ELECTRIC PANELS, BOXES, SWITCHES, RELAYS, WIRING		
50	PULLEYS, SPROCKETS, CHAINS, CABLES & Miscellaneous		34
51			
52			
53			
54			
55			
56	TOTALS - COCKPIT CONTROLS & AUTOPILOT	68	118
57	TOTAL (TO BE BROUGHT FORWARD)	186	

*From main distribution point to actuating units.

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SURFACE CONTROLS GROUP
SYSTEM CONTROLS

MODEL N-309
REPORT ____________

		All.	Elev.	Rud.	Wing Sweep	Wing Incid.	L.E. Flaps or Slats	T.E. Flaps	Spoilers	Speed Brakes	Stab. Adj.	Reaction Noz.
1												
2												
3												
4	CODE NO.											
5	MECHANICAL OPER. MECH.											
6	CONTROLS											
7	TENSION REGULATORS		10									
8	ACTUATORS											
9	TRIM CONTROLS											
10	Ratio Changers	10	10									
11	ELECTRICAL OPER. MECH.											
**12	TYPE						P					
13	CONTROLS											
*14	WIRINGS, SWITCHES, ETC.						9					
15	OPERATING MOTORS						4					
16	MECHANISM						15					
17	TRIM CONTROLS	3	3	3								
18	Stroke Limiter							2				
19	HYDRAULIC OPER. MECH.											
**20	TYPE	P	P	P				P				P
21	CONTROLS											
*22	PLUMBING & FLUID	12	10	9				2				
23	PUMPS											
24	RESERVOIRS											
25	ACCUMULATORS											
26	ACTUATORS	21	34	10				37				20
27	MECHANISM	27	28	16				5				10
28	TRIM CONTROLS											
29												
30	PNEUMATIC OPER. MECH.											
**31	TYPE											
32	CONTROLS											
*33	PLUMBING											
34	PUMPS											
35	BOTTLES (AIR)											
36	ACTUATORS											
37	MECHANISM											
38	TRIM CONTROLS											
39												
40	ARTIFICIAL FEEL	4	4	4								
41	BUNGEE											
42	BOB WEIGHT											
43												
44												
45												
46												
47												
48	SUPPORTS, GUIDES, ETC.											
49	WING											
50	TAIL											
51	BODY											
52	NACELLE											
53												
54	TOTALS	77	99	42			28	46				30
55	TOTALS - SYSTEM CONTROLS											322
56	TOTAL (FROM PG. 13)											186
57	TOTAL - SURFACE CONTROLS GROUP											508

*From main distribution point to actuating units.

**Type - add (P) for "Powered Controls"
- or (B) for "Boost Controls"

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ENGINE SECTION OR NACELLE GROUP

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		Lift Inboard Engines	Center	Cruise Outboard Engines	
1					
2					
3	CODE NO.				
4	ENGINE MOUNT - Engine	63		22	
5	- Diverter Valve Nozzles			26	
6	SUPPORT BAY				
7	VIBRATION ABSORPTION DEVICES				
8					
9					
10	NACELLE STRUCTURE			(332)	
11	BULKHEADS AND FRAMES			126	
12	COVERING & STIFFENERS			118	
13	FITTINGS				
14	LONGERONS			24	
15	ATTACHING ANGLES, ETC.			10	
16	Exhaust Blast Plates			54	
17					
18					
19	PYLONS & STRUTS				
20					
21					
22					
*23	FIREWALL & Shrouds	126		98	
24					
25	SHROUDS FOR FIRE PROTECTION				
26					
27	COWLING				
28	ENGINE COWL				
29					
30					
31					
32					
33					
34					
35	BAFFLES				
36	ACCESSORY COWL OR SKIRT				
37	COWL FLAPS				
38	COWL FLAP CONTROLS & OPERATING MECH.				
39					
40					
41					
42					
43					
44					
45	FAIRING - NACELLE TO WING OR PYLON				
46	STEPS & GRIPS				
47	WORKING PLATFORM (BUILT IN)				
48	INTERNAL WALKWAYS				
49					
50					
51	INSTALLATION HARDWARE				
52					
53					
54					
55					
56	TOTALS - SECTIONS OR NACELLES	189		478	
57	TOTAL (TO BE BROUGHT FORWARD)				667

*If in nacelle, or non-structural in wing or body.

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NACELLE GROUP
DOORS, PANELS & MISCELLANEOUS

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		*	**				Operating Mechanism			
		Location	Type Power	Area Sq. Ft.	Structure	Mechanism & Controls	Power Trans.	Actuator	Lock Mech.	Emerg.
1										
2										
3										
4	CODE NO.									
5	DOORS & FRAMES									
6	- LANDING									
7										
8										
9										
10	- BOMB									
11										
12										
13	- ACCESS	Nacelle		6.4	13					
14										
15	- ENGINE									
16										
17										
18										
19										
20										
21										
22										
23										
24										
25										
26	PANELS - (NON STRUCTURAL)									
27										
28										
29										
30										
31										
32										
33										
34										
35										
36										
37										
38										
39										
40										
41										
42										
43										
44										
45										
46										
47										
48										
49										
50										
51										
52	EXTERIOR FINISH									
53										
54	TOTALS				13					
55	TOTAL - DOORS, PANELS & MISC.									33
56	TOTAL FROM PG 15									667
57	TOTAL - ENGINE SECTION OR NACELLE GROUP									680

*Indicate location for major doors by inb'd, center, outb'd.

**H-Hydraulic, E-Electrical, P-Pneumatic; power transmission from main distribution point to actuating unit.

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PROPULSION GROUP

#	Item	Auxiliary - Lift		Main - Cruise	
2	CODE NO.				
3	ENGINE INSTALLATION		2940	X	934
4	ENGINE (AS INSTALLED)	2709		784	
5	ENGINE & AFTERBURNER (AS INSTALLED)				
6	REDUCTION GEAR BOX				
7	EXTENSION DRIVE SHAFT				
8	Diverter Valves			150	
9	Vectoring Nozzles	231			
10	AFTERBURNERS (IF FURNISHED SEPARATELY)				
11	ACCESSORY GEAR BOXES & DRIVES - CSD				140
12	SUPERCHARGER - COMPLETE (FOR TURBOS)				
13	- LUBRICATING SYSTEM				
14	- SUPPORTS				
15	- CONTROLS				
16	- PIPING (EXHAUST TO SUPER.)				
17					
18					
19	AIR INDUCTION SYSTEM		35	X	36
20	INTERCOOLERS AND SUPPORTS				
21	AIR DUCTING AND SHROUDING	35		36	
22	INTAKE DOORS & OPERATING CONTROLS				
23	AIR FILTERS				
24	SCREENS & CONTROLS				
25					
26					
27					
28					
29					
30					
31	EXHAUST SYSTEM		14	X	138
32	EXHAUST STACKS				
33	EXHAUST COLLECTORS				
34	COLLECTOR OR ENGINE SHROUD				
35	TAIL PIPE - Cruise Engines			44	
36	TAIL PIPE ~~XXXXXXXXXXXXXX~~ - Diverters			80	
37	TAIL CONE				
38	SILENCING DEVICES				
39	SUPPORTS, BRACKETS, ETC.				
40	Ejector Pumps	14		14	
41					
42					
43					
44	COOLING SYSTEM			X	
45	RADIATOR AND SUPPORTS				
46	SHUTTERS, SCOOP & DUCTS				
47	EXPANSION TANK & SUPPORTS				
48	LIQUID IN SYSTEM(GALS.)				
49	PIPING, VENTS, CLAMPS ETC.				
50					
51					
52	FANS				
53	CONTRAVANES				
54	FAN DRIVES				
55	CONTROLS & OPERATING MECH.				
56					
57	TOTALS (TO BE BROUGHT FORWARD)		2989	X	1248

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PROPULSION GROUP
LUBRICATING & FUEL SYSTEMS

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Line	Item	Type	Location	No.	Vol. (Gals.)	Auxiliary -Lift Lubricating	Auxiliary -Lift Fuel	Main-Cruise Lubricating	Main-Cruise Fuel
1									
2									
3					CODE NO.				
4	TANKS	Type	Location	No.	Vol. (Gals.)				
5		Bladder	Fwd. Fus.	2	276.9				34
6		Bladder	Center Fus.	2	107.7				19
7		Bladder	Aft Fus.	2	323.1				37
8									
9									
10									
11									
12									
13									
14									
15									
16									
17									
18									
19									
20	INTEGRAL TANK SEALS & SEALANT								
21	BACKING BOARD								
22	TANK SUPPORTS & PADDING								55
23	TANK BAY SEALING								
24	Thermal Insulation								15
25	TANK RELEASE & CONTROLS								
26	OIL COOLING INSTALLATION								
27	COOLERS & SUPPORTS (SIZE) (NO.)								
28	DUCTS & SHUTTERS								
29	AUTOMATIC OIL TEMP. CONTROL VALVE								
30	SHUTTER CONTROLS								
31									
32	FUEL VAPOR RECOVERY								
33									
34	OIL DILUTION SYSTEM								
35									
36	FUEL VAPOR INERTION SYSTEM - CYL. & SUPPORTS								
37	- GENERATOR								
38	- CONTROLS, ETC.								
39	PUMP INSTALLATION	Type		No.					
40	ENGINE DRIVEN								
41	BOOSTER	Vane		4					28
42	HAND (INCL. CONTROLS)								
43	TRANSFER								
44									
45									
46	FILLING SYSTEM - GROUND					7		2	12
47	- IN FLIGHT								
48									
49	DISTRIBUTION SYSTEM								143
50	TRANSFER SYSTEM								6
51	VENT SYSTEM								15
52	PRESSURIZATION SYSTEM								
53	DUMP SYSTEM								
54	Drains								11
55									
56	TOTALS - LUBRICATING & FUEL SYSTEMS					7		2	375
57	TOTALS (TO BE BROUGHT FORWARD)					7		377	

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PROPULSION GROUP

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		Type	*Aux.	*Main	Code No.	Auxiliary-Lift		Main-Cruise	
1									
2					CODE NO.				
3	WATER INJECTION SYSTEM								
4	TANKS (NO.) (GALS/TANK)								
5	PUMP								
6	METERING UNIT								
7	VALVES & PLUMBING								
8	CONTROLS								
9									
10									
11	ENGINE CONTROLS						50		28
12	IGNITION								
13	THROTTLE Quadrants							15	
14	MIXTURE								
15	SUPERCHARGER (SUP. INTEG. WITH ENG.)								
16	AFTERBURNER								
17	Amplifiers					10		3	
18	Actuators					28		4	
19	Installation					12		6	
20	STARTING SYSTEM - Air Impingement						26		15
21	STARTER POWER UNIT (TYPE:)								
22	STARTER (TYPE:)								
23	STARTER CONTROLS								
24	CRANK & EXTENSION								
25	PRIMER & PIPING								
26	MESHING SOLENOID								
27	SWITCHES, WIRING & CONDUIT								
28	Valves					19		10	
29	Ducts					7		5	
30									
31									
32	PROPELLER INSTALLATION (DIA.) (NO.)								
33	PROPELLER								
34	CUFFS								
35	SPINNER								
36	CONTROLS	Type	*Aux.	*Main					
37	SPEED								
38	PITCH								
39	FEATHERING								
40	REVERSING								
41									
42									
43									
44									
45									
46									
**47	OIL (GALS)								
**48	TANK & PLUMBING								
49									
50									
51									
52									
53									
54									
55	TOTALS						76		43
56	TOTALS FROM PGS. 17 & 18						2996		1625
57	TOTAL - PROPULSION GROUP							4740	

*GFP Weight

**When separate oil system used.

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AUXILIARY POWER PLANT GROUP

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		CODE NO.
1		
2	ENGINE OR POWER UNIT (MODEL)	
3	ENGINE SUPPORTS	
4		
5		
6	AIR INDUCTION SYSTEM	
7		
8		
9	EXHAUST SYSTEM	
10	STACKS	
11	COLLECTOR	
12		
13		
14		
15	COOLING SYSTEM	
16	RADIATORS	
17		
18		
19		
20	LUBRICATING SYSTEM	
21	TANKS (VOL. GALS.)	
22	PLUMBING	
23		
24		
25		
26	FUEL SYSTEM	
27	TANKS (VOL. GALS.)	
28	PLUMBING	
29		
30		
31	CONTROLS	
32		
33		
34		
35	STARTING SYSTEM	
36	STARTER	
37		
38		
39		
40	SUPPORTS	
41		
42		
43		
44		
45		
46		
47		
48		
49		
50		
51		
52		
53		
54		
55		
56		
57	TOTAL - AUXILIARY POWER PLANT GROUP	0

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INSTRUMENT & NAVIGATIONAL EQUIPMENT GROUP
INSTRUMENTS

	* FUNCTIONAL GROUPS & ITEMS	Number	Indicator	Transmitter & Amplifier	Installation	Total
1						
2						
3	CODE NO.					
4						
5	Fwd Cockpit - Flight Instruments		(17.2)	(0.3)	(1.0)	(18.5)
6	Airspeed/Machmeter	1	1.8		0.8	2.6
7	Clock	1	0.4			0.4
8	Standby Compass	1	0.9			0.9
9	Vertical Velocity	1	1.7			1.7
10	Pressure Altimeter	1	1.8			1.8
11	Flap Position	1	0.4	0.3	0.2	0.9
12	Turn, Slip & Side Slip	1	1.2			1.2
13	Attitude	1	3.0			3.0
14	Bearing	1	3.1			3.1
15	Take-off Trim	1	0.6			0.6
16	Angle of Attack	1	0.8			0.8
17	Radar Altimeter	1	1.5			1.5
18						
19	Fwd Cockpit - Engine Instruments		(28.8)	(24.0)	(1.7)	(54.5)
20	Oil Pressure - Cruise Engine	2	2.0	3.0		5.0
21	Fuel Management		4.5	16.5		21.0
22	Tachometers - Cruise Engine	2	1.8			1.8
23	Exhaust Gas Temp. - Cruise Engine	2	3.0			3.0
24	Hydraulic Pressure	2	1.0	3.0	1.7	5.7
25	Multiple Thrust Indication	1	10.0			10.0
26	Thrust Vector	1	1.5	1.5		3.0
27	Lift Engine Start Lights (RPM,EGT,Oil)	8	5.0			5.0
28						
29	Aft Cockpit - Flight Instruments		(15.6)	(0.3)	(1.0)	(16.9)
30	Accelerometer	1	0.8			0.8
31	Airspeed/Machmeter	1	1.8		0.8	2.6
32	Clock	1	0.4			0.4
33	Vertical Velocity	1	1.7			1.7
34	Pressure Altimeter	1	1.8			1.8
35	Flap Position	1	0.4	0.3	0.2	0.9
36	Turn, Slip & Sideslip	1	1.2			1.2
37	Attitude	1	3.0			3.0
38	Bearing	1	3.1			3.1
39	Take-off Trim	1	0.6			0.6
40	Angle of Attack	1	0.8			0.8
41						
42	Aft Cockpit - Engine Instruments		(28.2)	(0)	(0)	(28.2)
43	Oil Pressure - Cruise Engine	2	2.0			2.0
44	Fuel Flow - Cruise Engine	2	2.0			2.0
45	Tachometer - Cruise Engine	2	1.8			1.8
46	Exhaust Gas Temp. - Cruise Engine	2	3.0			3.0
47	Hydraulic Pressure	2	1.0			1.0
48	Fuel Quantity - Repeat Ind.	1	0.9			0.9
49	Multiple Thrust Indication	1	10.0			10.0
50	Thrust Vector	1	1.5			1.5
51	Lift Engine Start Lights	8	5.0			5.0
52	Strut Pressure Main & Nose Gears	1	1.0			1.0
53	INSTRUMENT POWER SYSTEM (TYPE					
54	Wiring & Misc. Installations				52.9	52.9
55						
56						
57	TOTAL - INSTRUMENTS (TO BE BROUGHT FORWARD)					171.0

*List items by functional groups (Flight, Engine & Misc.). List sub-groups by crew stations; add supp. pg. 21A if necessary.

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INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP
NAVIGATIONAL

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		CODE NO.	Installation	Navig. Equip.
1				
2				
3	NAVIGATIONAL. INSTALLATION			
4	LOCKERS & SUPPORTS			
5	TABLES & SUPPORTS			
6	SHELVES & SUPPORTS			
7	MAP CASES & REPORT HOLDERS			
8	ASTRODOME			
9	CHARTBOARD			
10	STOWAGES & SUPPORTS - CHARTBOARD			
11	- SPOTLIGHT			
12	- DRIFTSIGHT			
13	- ASTROCOMPASS			
14				
15				
16				
17				
18				
19				
20				
21				
22				
23				
24				
25	NAVIGATIONAL EQUIPMENT			
26	PLOTTER			
27	PARALLEL RULER			
28	CHARTS			
29	DRIFT SIGHT			
30	SEXTANT & CASE			
31	BINOCULARS			
32	NAVIGATING WATCHES & BOXES			
33	COMPUTERS			
34	ASTROCOMPASS & ADAPTERS			
35				
36				
37				
38				
39				
40				
41				
42				
43				
44				
45				
46				
47				
48				
49				
50				
51				
52				
53				
54	TOTALS			
55	TOTAL - NAVIGATIONAL		0	
56	TOTAL - INSTRUMENTS FROM PG. 21		171	
57	TOTAL - INSTRUMENT & NAVIGATIONAL EQUIP. GROUP			171

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*HYDRAULIC & PNEUMATIC GROUP

Item	Code No.	Hydraulic Utility	Hydraulic Emergency		Pneumatic Utility	Pneumatic Emergency	
PUMPS & COMPRESSORS	Model						
CSD Mounted	2 Req.	28					
Engine Mounted	2 Req.	28					
REMOTE PUMP DRIVES							
(No. / Model / Capacity)							
RESERVOIRS — No. 2, Capacity 1 gal.		20					
AIR BOTTLES							
ACCUMULATORS							
FILTERS		12					
PRESSURE REGULATORS		2					
Low Pressure Warning		1					
VALVES -		2					
Manifolds		4					
Heat Exchanger		5					
CONTROLS							
PLUMBING & Miscellaneous		70					
FLUID IN SYSTEM (TYPE) (GALS.)		18					
SUPPORTS - WING							
- TAIL							
- BODY							
- NACELLE							
TOTALS		190					
TOTAL - HYDRAULIC & PNEUMATIC GROUP							190
FURNISHES POWER FOR - (ITEMS)		Actuation of Control Surfaces					
		Alighting Gear					
		Brakes					
		SAS Actuators					
		VSS Actuators					
		Engine Doors					
SYSTEM PRESSURE (PSI)							

*Includes system from sources of power to main distribution points.

HOVER CONTROL SYSTEM

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[redacted]

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[redacted]

	Engine Bleed	Forward Fuselage	Center Fuselage	Aft Fuselage	Wing*
Ducts	16	24	49	24	34
Transition Fittings	18		16		2
Bellows	37	13	17	13	9
Joints - Bolted Flanges	8	8	16	8	10
- Clamped Couplings	20				2
Insulation		14		14	7
Valves - Check	27				
- Reaction Control		24		24	19
Tip Pods					30
Fairings				25	
Totals	126	83	98	108	113
Total Hover Control System					528

Add supplementary pg. iiA if necessary.

*Includes wing take-off ducting in center fuselage.

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*ELECTRICAL GROUP

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						AC		DC	
1									
2									
3				CODE NO.					
4	POWER SUPPLY EQUIPMENT	Driven By	KVA	Amp.	No.		76		16
5	GENERATORS	CSD	15		2	74			
6									
7									
8									
9									
10	REMOTE GENERATOR DRIVES- CSD in Propulsion Group								
11	BATTERY (AN) (NO. 1) 24 V., 5 A.H.							15	
12	BATTERY CONTAINER, OVERFLOW INST. & SUPPORTS							1	
13	Generator Cooling Ducts					2			
14	POWER CONVERSION EQUIPMENT	Model			No.		13		18
15	INVERTER (DC TO AC)- Static				1	13			
16	CONVERTER (AC TO DC)								
17	TRANSFORMER/Rectifiers				2			18	
18	RECTIFIER								
19	MOTOR-GENERATOR								
20	PHASE ADAPTER								
21	FREQUENCY CONVERTER								
22									
23									
24	POWER DISTRIBUTION & CONTROL						156		13
25	GENERATOR CONTROL BOXES					15			
26	CUTOUTS & VOLTAGE REGULATORS					6			
27	AMMETERS & VOLTMETERS								
28	SWITCHES, RHEOSTATS, SWITCH PANELS Autoxformers					15			
29	CIRCUIT BREAKERS & FUSES Panels					10		7	
30	JUNCTION, FUSE, DISTRIBUTION BOXES & PANELS					7			
31	RECEPTACLES & CONNECTOR PLUGS								
32	RELAYS (Power Contactors)					5		1	
33	WIRING & Connectors					75		5	
34	CONDUIT					15			
35	Current Transformers (Power Control)					8			
36	LIGHTS & SIGNAL DEVICES						23		
37	LIGHTS - INTERIOR					4			
38	- EXTERIOR					12			
39	- LANDING (INCL. RETRACT MECH.)					--			
40									
41	SIGNAL DEVICES - LIGHTS					6			
42	- HORNS					1			
43	- BELLS								
44									
45									
46	EQUIPMENT SUPPORTS - WING						20		
47	- TAIL					20			
48	- BODY								
49	- NACELLE								
50	TOTALS						288		47
51	TOTAL - ELECTRICAL GROUP					335			
52	FURNISHES POWER FOR - (ITEMS)					Electronics,		Fire Detection,	
53						Lights, Instru-		Electronics,	
54						ments, Fuel		Lights, Genera-	
55						System, Ignition,		tor Control	
56						Flap, Misc. Sys-		& Instruments	
57						tems, Controls			
						and D.C. System			

*Includes system from sources of power to main distribution points.

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ELECTRONICS GROUP

MODEL N-309
REPORT

*FUNCTIONAL GROUP	*EQUIPMENT COMPONENTS & PART NUMBERS OR IDENTIFICATION	CODE NO.	Equipment GFP	Equipment CFE	Installation
VORTAC			(65)		
	AN/RNA-26C VHF Nav. Rec.		25		
	DME-DMA-296		29		
	Control Panels (2)		6		
	Antennas		5		
IFF/SIF			(17)		
	AN/APX-72 Transponder		12		
	Control Panels (2)		4		
	Antennas		1		
ILS			(10)		
	MKA-28 Marker Beacon		4		
	Antenna		2		
	Controls (2)		4		
Communication			(29)		
	RTA-41B VHF Transceiver		23		
	Control Panels (2)		6		
Intercom			(9)		
	Control Boxes (2)		5		
	Microphone Amplifiers (2)		2		
	Headset Amplifiers (2)		2		
Compass System	- Gyro Stabilized		(29)		
	Stable Platform, SR-3		21		
	Heading Coupler		5		
	Controllers (2)		3		
	Indicators (In Instruments Group)		--		
Miscellaneous				(15)	(46)
	Radio "J" Box			12	
	Noise Filters			2	
	Antenna Switching Relays			1	
	Wiring, Coax & Misc.				46
ELECTRONIC INSTALLATION					
TABLES					
RACK, SHELVES & SUPPORTS					
LOCKERS					
SUBTOTALS - EQUIPMENT GFP & CFE			159	15	
TOTALS			174		46
TOTAL - ELECTRONIC GROUP					220

*List components (incl. Radomes, Mts., Antennae, Switches, Relays, Filters, etc.) from main distribution point to unit operated, by functional groups (e.g., Comm., VHF, Search, Navig., Intercomm., etc.). Add supplementary pg. 25A if necessary.

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ARMAMENT GROUP

PAGE 33
MODEL N-309
REPORT

1	TURRET OR GUN LOCATIONS						
2	NO. OF GUNS						
3	CALIBER OF GUNS						
4	CODE NO.						
5	EQUIPMENT						
6	AMMUNITION - FEED CHUTES						
7	- BOXES						
8	- BOX SUPPORTS						
9	CASE EJECTION CHUTES						
10	LINK EJECTION CHUTES						
11	CASE & LINK RECEPTACLES						
12	MOUNTS						
13	MOUNT SUPPORTS						
14	BLAST TUNNELS (INTEGRAL)						
15							
16	AMMUNITION BOOSTERS						
17							
18	GUN CHARGERS INSTALLATION						
19							
20							
21							
22							
23	HEATING						
24							
25	FIRE INTERRUPTERS						
26							
*27	TURRET SYSTEMS (TYPE:)						
28							
28							
30							
31							
32							
33							
34							
35							
36							
37							
38							
39							
40							
41							
42							
43							
44							
45							
46							
47							
48							
49							
50							
51							
52							
53							
54							
55							
56	TOTALS						
57	TOTAL (TO BE BROUGHT FORWARD)						0

*List components by packaged groups (turret structure, accessories, sighting, computation, gun accessories, elec. cabling, etc.). Add supplementary pg. 26A if necessary.

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ARMAMENT GROUP

PAGE 34

MODEL N-309

REPORT ____________

		CODE NO.	
1			
2	ROCKET INSTALLATION - CONTROLS		
3	- RELEASE MECHANISM		
*4	- WIRING, ETC.		
5			
6			
7			
8			
9			
10			
11	BOMB, MINE & TORPEDO INSTALLATION		
12	BOMBER'S CONTROL PANEL		
*13	RELEASE - ELECTRICAL		
14	- MANUAL		
15	- MANUAL EMERGENCY		
16	INTERVALOMETER		
17			
18	ARMING & SAFE MECHANISM		
19			
20	BOMB HOIST MECHANISM (IF BUILT IN)		
21			
**22	BOMB RACKS (TYPE) (NO.)		
23			
24	BOMB RAILS OR TRACKS		
25			
26			
27			
28			
29			
30	GUNFIRE PROTECTION (PASSIVE DEFENSE)		
31	BULLET RESISTANT GLASS		
32	ARMOR PLATE (NON STRUCTURAL) - OIL COOLER		
33	- OIL TANK		
34	- ENGINE SECTION		
35	- CREW - PILOT		
36	-		
37	-		
38	-		
39			
40			
41			
42	FLAK CURTAINS		
43			
44			
45	SONOBUOY - STOWAGE		
46	- DISPENSERS		
47			
48			
49			
50			
51			
52			
53	SMOKE TANK PROVISIONS		
54			
55	TOTAL		
56	TOTAL FROM PG. 26		
57	TOTAL - ARMAMENT GROUP		0

*From main distribution point to actuating unit.

**If not specified as useful load.

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FURNISHINGS & EQUIPMENT GROUP
ACCOMMODATIONS FOR PERSONNEL

PAGE 35
MODEL N-309
REPORT ______

1	CODE NO.									
2	CREW SEATS & PASSENGER CHAIRS									
3–4	Location	No.	Surv.Kit Container	Seat	Safety Belt	Harness & Inertia Reel	Adj. Mech.	Catapult or Eject. Mech.	Tracks & Supports	
5	PILOT	1	20	69	2	3	7	24	15	140
6	ASST. PILOT	1	20	69	2	3	7	24	15	140
7										
8										
9										
10										
11										
12										
13										
14										
15										
16										

17	HEAD REST (IF NOT INTEGRAL WITH SEAT)	
18	BUNKS (NO.) & SUPPORTS	
19		
20	LITTER SUPPORTS	
21	BOMBERS & GUNNERS KNEELING PADS (NO.)	
22	PARACHUTE STOWAGE PROVISIONS	
23	TOILETS & RELIEF TUBES	
24	WASH BASIN & SHOWERS	
25	WATER TANKS & PIPING	
26	DRINKING WATER CONTAINERS & SUPPORTS	
27	LOCKERS FOR - FOOD	
28	- PERSONAL EFFECTS	
29		
30	GALLEY STOVES & HOTPLATES	
31	REFRIGERATOR	
32		
33		
34		
35		
36		
37	ANTI-G SUIT PROVISIONS	
38		
39	OXYGEN INSTALLATION (Gaseous)	(25)
*40	BOTTLES - INCL. CHARGE (TYPE 1800 psi) (SIZE 22 cu.ft.) (NO. 1)	12
41	(at atmospheric pressure)	
42	CONVERTOR	
*43	REGULATORS (TYPE) (NO. 2)	5
44	SUPPORTS - BOTTLES & REGULATORS	
45	PLUMBING, ETC.	8
46		
47		
48		
49		
50		
51		
52		
53		
54		
55		
56		
57	TOTAL - ACCOMMODATIONS FOR PERSONNEL (TO BE BROUGHT FORWARD)	305

*If not specified as useful load or special equipment.

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FURNISHINGS & EQUIPMENT GROUP
MISC. EQUIPMENT & FURNISHINGS

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MODEL N-309
REPORT ____________

Line	Item	CODE NO.	Misc. Equip.	Furnishings
1				
2				
3				
4	MISCELLANEOUS EQUIPMENT			
* 5	PORTABLE PLATFORMS & LADDERS			
6				
7	DATA CASES & REPORT OR FORM HOLDERS			4
8				
* 9	MANUALS - FLIGHT & MAINTENANCE -			
*10	BALANCE COMPUTER & SUPPORT			
11				
12	TOOL LOCKERS			
13				
14	WINDSHIELD WIPER & WASHER INSTALLATION			
15	RELEASE MECHANISM & FITTINGS - TARGET & GLIDER TOW			
16				
17	BILGE SYSTEM			
18	STALL WARNING DEVICES			
19	REAR VIEW MIRROR		1	
20	Instrument Panel Glare Shields		4	
21	AUXILIARY FLOORING			
22	INSTRUMENT BOARDS & Supports		16	
23	CONSOLES & Pedestals		23	
24	CONTROL STANDS			
25				
*26	CARGO HANDLING EQUIPMENT			
27	RAMPS			
28	HOISTS & BOOMS			
29	MONORAILS			
30	MONORAIL MOTORS			
31	TIE DOWN FITTINGS			
32				
33				
34				
35	PYROTECHNIC INSTALLATION			
36	SIGNAL PISTOL HOLDER			
37	SIGNAL AMMUNITION HOLDER (CAP.)			
38	PARACHUTE FLARE - CONTAINERS (NO.)			
39	- RACKS (CAPACITY)			
40	- RELEASE MECHANISM			
41				
42	SMOKE CANDLE (GRENADE) HANDLE			
43				
44	FLOAT LIGHT RACK & RELEASE MECH. (CAP.)			
45				
46	FURNISHINGS			
47	FLOOR COVERING, RUGS, ETC.			
48	SOUNDPROOFING & THERMAL INSULATION			17
49	TRIM			
50	CURTAINS & SCREENS			
51	CRASH PADDING			
52	PARTITIONS (NON-STRUCTURAL)			
53	Miscellaneous Stowage Provisions			5
54				
55				
56	TOTALS - MISC. EQUIP. & FURNISHINGS		44	26
57	TOTAL (TO BE BROUGHT FORWARD)		70	

*If not specified as special equipment.

FURNISHINGS & EQUIPMENT GROUP
EMERGENCY EQUIPMENT

1		CODE NO.							
2			Compartments						
3	FIRE EXTINGUISHERS		Engine	Baggage	Fuel				
4	BOTTLES	Type							
5		Size							
6		No.							
7		Weight							
8	CONTROLS								
9	PLUMBING								
10	BOTTLE SUPPORTS								
11									
12									
13	TOTAL - COMPT.								
14	PORTABLE (TYPE) (SIZE) (NO.)								
15									
16									
17									
18	PORTABLE EXTINGUISHER SUPPORTS								
19									
20	FIRE DETECTION SYSTEM - Lift Engines (7)								25
21	- Cruise Engines (2)								10
22	FIRE RESISTANT PAINT								
23	FIRE CURTAINS								
24									
25	FIRST AID KITS (NO.) & STOWAGE								
26									
27	FLASHLIGHTS (NO.)								
28									
29	STOWAGE - EMERGENCY RATIONS & WATER								
30									
*31	LIFE RAFTS - (TYPE) (NO.)								
32									
33									
34									
35									
36	- SUPPORTS OR CRADLES								
37									
38	DITCHING STATION EQUIPMENT								
39									
40									
41									
42									
43									
44									
45									
46									
47									
48									
49									
50									
51									
52									
53									
54									
55	TOTAL - EMERGENCY EQUIPMENT								35
56	TOTALS FROM PGS. 28 & 29								375
57	TOTAL - FURNISHINGS & EQUIPMENT GROUP								410

*If not specified as useful load or special equipment.

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AIR CONDITIONING & ANTI-ICING
EQUIPMENT GROUP
AIR CONDITIONING

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REPORT

	Pressurization System	Ventilating System	Heating System	Cabin Cooling System	Equip. Cooling	
1						
2						
3 CODE NO.						
* 4 HEAT EXCHANGERS (NO.)				25		
5						
* 6 HEATERS (BTU CAPACITY) (NO.)						
7						
8 HEATING FLUID (GALS.)						
9						
10 COMPRESSORS OR SUPERCHARGERS						
11						
12 MOTORS						
13 TURBINES						
14 FANS & Blowers					4	
15						
16						
17 TANKS						
18 WATER SEPARATOR			2			
19 REGULATOR						
20						
21						
22 SCOOPS & Ram Air Valves					4	
23 DUCTING			3	16		
24 SHROUDS						
25						
26						
27 PLUMBING						
28						
29						
30 BOMB BAY HEATING						
31						
32						
33						
34						
35						
36 CONTROLS - MANUAL						
37						
38 - ELECTRICAL				7		
39						
40 - HYDRAULIC						
41						
42 - PNEUMATIC						
43						
44						
45						
46						
47						
48 SUPPORTS & BRACKETS - WING						
49 - TAIL						
50 - BODY						
51 - NACELLE						
52						
53 PRESSURIZATION SEALING						
54						
55						
56 TOTALS			5	48	8	
57 TOTAL - AIR CONDITIONING (TO BE BROUGHT FORWARD)						61

*If not specified as special equipment.

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AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP
ANTI-ICING

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MODEL N-309
REPORT ______

Line	Item	No.	BTU Capacity	Wing	Tail	Air Induction	Propeller	Canopy & Windshield	Fuel System	
1										
2										
3			CODE NO.							
* 4	HEATERS	No.	BTU Capacity							
5										
6										
7										
8										
9										
10										
*11	HEAT EXCHANGERS (NO.)									
12										
13										
14										
15	DUCTING							4		
16	SHROUDING									
17										
18										
*19	BOOTS									
20										
*21	ATTACHING STRIPS									
22										
23	OIL SEPARATORS									
24										
25	AIR PUMPS									
26										
27	AIR LINES & HOSES									
28										
29	TANKS									
30										
*31	FLUID (GALS.)									
32										
33										
34										
35	PLUMBING									
36										
37										
38	DISTRIBUTOR - VALVE									
39	- CONTROLS									
40										
41										
42	CONTROLS - MANUAL							2		
43	- ELECTRICAL									
44	- HYDRAULIC									
45	- PNEUMATIC									
46										
**47	WIRING, SWITCHES, RELAYS									
48										
49	SUPPORTS & BRACKETS - WING									
50	- TAIL									
51	- BODY							1		
52	- NACELLE									
53										
54	TOTALS							7		
55	TOTAL - ANTI-ICING									7
56	TOTAL FROM PG. 31									61
57	TOTAL - AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP									68

*If not specified as special equipment.

**From main distribution point to actuating unit.

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PHOTOGRAPHIC GROUP

PAGE 40
MODEL N-309
REPORT

				Photo Equipment	Installation Provision
1					
2					
3			CODE NO.		
* 4	EQUIPMENT ITEM	Model			
5	CAMERAS				
6					
7					
8					
9					
10					
11					
12					
13					
14					
15					
16					
17					
18					
19					
20					
21	MOUNTS & ADAPTERS				
22					
23					
24					
25					
26					
27					
28	INSTALLATION PROVISIONS				
29	TRIP & SYNCHRONIZATION UNITS				
30					
31					
32					
33	PHOTO-ELECTRIC SYSTEM				
34					
35					
36					
37	FLASHLIGHT BOMB INST.				
38					
39					
40					
41	VACUUM SYSTEM - PUMP				
42	- PLUMBING				
43	- GAGE				
44					
45					
46	PRESSURE SYSTEM - PUMP				
47	- PLUMBING				
48	- GAGE				
49					
50	CONTROLS				
51	SUPPORTS				
52					
**53	WIRING, RELAYS, SWITCHES				
54					
55					
56	TOTALS				
57	TOTAL - PHOTOGRAPHIC GROUP				0

*If not specified as useful load or special equipment.

**From main distribution point to actuating unit.

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NAME ______________
DATE ______________

AUXILIARY GEAR GROUP

MODEL N-309
REPORT ______________

		Handling	Arrest.	Catapult	ATO
1					
2	CODE NO.				
3	HANDLING GEAR				
4	ANCHOR				
5	ANCHOR LINE				
6	PENDANT & CLAMP FITTING				
7	LIZARD				
8	SHEAVES				
9	WINCH - COMPLETE				
10	WINCH CRANK				
11	ANCHOR HANDLING RIG OR DAVIT				
12	WINCH ENGINE OR MOTOR				
13					
*14	HOISTING SLING				
15	WING HANDLING LINES				
16	WATER RUDDER				
17	FITTINGS - RECOVERY HOOK				
18	- BEACHING GEAR ATTACHMENT				
19	- TIEDOWN				
20	- JACKING				
21	- TOWING				
22	- MOORING & SNUBBING				
23	- ANCHORAGE				
24	- LEVELING				
25	- HOISTING				
26					
27	ARRESTING OR DECELERATION GEAR				
28	TRAILING HOOK				
29	HOOK POINT (TYPE)				
30	EXTENSION GEAR				
31	RETRIEVING GEAR				
32	BUMPER				
33	SHOCK ABSORBER				
34	ATTACHMENT FITTINGS				
35					
36	BARRIER CRASH FITTINGS				
37					
38	DECELERATION - PARACHUTE				
39	- CONTAINER & FITTINGS				
40	- CONTROLS				
41					
42					
43	CATAPULTING GEAR				
44	CATAPULT FITTINGS				
45	CATAPULT HOOKS				
46	HOLDBACK FITTINGS				
47					
48	ASSISTED TAKE OFF				
49	HOOKS				
50					
51	CONTROLS - FIRING				
52	- BOTTLE RELEASE				
53	BOTTLE STOWAGE PROV. (NO. BOTTLES)				
54					
55					
56	TOTALS				
57	TOTAL - AUXILIARY GEAR GROUP				0

*If not specified as special equipment.

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FORM 20-7A
(R.11-63)

ENGINEER	NORTHROP CORPORATION NORAIR DIVISION	PAGE 42
CHECKER		REPORT NO.
DATE	Variable Stability System (Not Included in Weight Empty)	MODEL

	Equipment	Installation	Total
Pitot Boom & Mount	14		14
q Sensor	3		3
Angle of Attack & Sideslip Sensors	5		5
Aeroflex Sensors	7		7
Aeroflex Unit & Indicator	15		15
3 Axis Accelerometers - Angular	2		2
- Translation	1		1
Attitude Gyro	7		7
Airborne Computer	74		74
Autopilot Unit	18		18
Cockpit Controllers	8		8
Transducers	16		16
Parallel Actuators	21		21
Radar Altimeter - Rec/Xmitter	10		10
- Antennas	2		2
Doppler Radar - R-T Unit & Antenna	21		21
- Signal Converter	17		17
- Cockpit Controller (1)	4		4
- Mounting Rack		2	2
Provisions For Future Units	86	43	129
Installation - Electronic Units		62	62
- Servo/Mechanical		62	62
Totals	331	169	
Total Variable Stability System			500

The following Airframe Weight Derivation is based on the Aeronautical Manufacturer's Planning Report DD387-1 Form dated June 1958.

		Weight Lbs.
N-309 - AD-4486A - Weight Empty - Composite Mode		12,685
Less:		
1. Wheels, Brakes, Tires	196	
2. Engines & Diverters	3,874	
3. Cooling Fluid	0	
4. Fuel Cells	90	
5. Starters	0	
6. Propellers	0	
7. APU	0	
8. Instruments	117	
9. Navigational Equipment (incl. in #11)	0	
10. Batteries, Electrical Power Supply, & Conversion Equipment	120	
11. Electronics	159	
12. Turrets	0	
13. Air Conditioning	25	
14. Cameras	0	
Contingency	267	
N-309 - AD-4486A - AMPR Weight		7,837

FORM 20-23A (R.8-59)

NORTHROP CORPORATION
NORAIR DIVISION

Empty WEIGHT AND BALANCE

Composite Mode
GROUP OR CONDITION

ENGINEER
CHECKED
DATE

PAGE NO. 44
MODEL N-309
REPORT NO.

ITEM ($\bar{c}$ = 75.3, LEMAC = 374.5)	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ. DIST.	HORIZ. MOMENT 2	HORIZ. MOMENT 1	VERT. DIST.	VERT. MOMENT 2	VERT. MOMENT 1
Wing Group			930	407.0		378,510	100.0		93,000
Tail Group			309	589.2		182,070	130.9		40,448
Body Group			2,537	390.6		990,855	109.9		278,860
Alighting Gear Group (Down)			792	376.1		297,864	60.0		47,520
Surface Controls Group			508	380.6		193,366	103.5		52,603
Engine Section			680	415.4		282,484	121.4		82,579
Propulsion Group			4,740	385.6		1,827,696	117.3		556,120
Instruments Group			171	270.5		46,259	120.6		20,618
Hydraulics Group			190	419.0		79,614	111.7		21,232
Electrical Group			335	410.6		137,554	113.6		38,064
Electronics Group			220	259.1		56,997	106.2		23,372
Furnishings & Equipment Group			410	245.3		100,581	115.9		47,509
Air Conditioning Group			68	315.7		21,470	97.6		6,640
Hover Controls Pneumatic System			528	410.0		216,492	97.6		51,552
Contingency			267	388.0		103,596	110.0		29,370
Weight Empty (Gear Down)			12,685	387.5 17.3% MAC		4,915,408	109.5		1,389,487
Retract Gear						310			28,235
Weight Empty (Gear Up)			12,685	387.5 17.3% MAC		4,915,718	111.8		1,417,722

NORTHROP CORPORATION
NORAIR DIVISION

WEIGHT AND BALANCE

Composite Mode
GROUP OR CONDITION

ENGINEER
CHECKED
DATE

MODEL N-309
REPORT NO.

ITEM ($\bar{c}$ = 75.3, LEMAC = 374.5)	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ. DIST.	HORIZ. MOMENT 2	HORIZ. MOMENT 1	VERT. DIST.	VERT. MOMENT 2	VERT. MOMENT 1
Weight Empty (Gear Down)			12,685	387.5		4,915,408	109.5		1,389,487
Crew			400	231.0		92,400	126.0		50,400
Pilot		200		202.0	40,400		124.0	24,800	
Safety Pilot		200		260.0	52,000		128.0	25,600	
Engine Oil			63	383.0		24,129	122.0		7,686
Unusable Fuel			45	390.0		17,550	106.0		4,770
Operating Weight Empty (Gear Down)			13,193	382.7		5,049,487	110.1		1,452,343
Retract Gear						310			28,235
Operating Weight Empty (Gear Up)			13,193	382.8		5,049,797	112.2		1,480,578
Payload			800	426.7		341,327	108.2		86,591
NASA Research Equipment		300		547.0	164,100		120.0	36,000	
Variable Stability System		500		354.5	177,227		101.2	50,591	
Gross Weight Less Fuel (Gear Up)			13,993	385.3 14.3% MAC		5,391,124	112.0		1,567,169
Extend Gear						-310			-28,235
Gross Weight Less Fuel (Gear Down)			13,993	385.3 14.3% MAC		5,390,814	110.0		1,538,934
Fuel			4,007	396.7		1,589,388	114.9		460,226
Forward Fuel		1,800		292.0	525,600		118.0	212,400	
Aft Fuel		2,100		487.0	1,022,700		112.0	235,200	
Center Fuel		107		384.0	41,088		118.0	12,626	
VTOL (Gear Down)			18,000	387.8 17.7% MAC		6,980,202	111.1		1,999,160
Retract Gear						310			28,235
VTOL (Gear Up)			18,000	387.8 17.7% MAC		6,980,512	112.6		2,027,395

Form 20-23A (R.8-59)

NORTHROP CORPORATION
NORAIR DIVISION

WEIGHT AND BALANCE

Derivation of Alighting Gear Retraction Moment
GROUP OR CONDITION

ENGINEER
CHECKED
DATE

MODEL N-309
REPORT NO.

ITEM	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ. DIST.	HORIZ. MOMENT 2	HORIZ. MOMENT 1	VERT. DIST.	VERT. MOMENT 2	VERT. MOMENT 1
Alighting Gear (Down)			792	376.1		297,864	60.0		47,520
Remove:									
Nose Gear Down		-185		217.0	- 40,145		60.0	-11,100	
Main Gear Down		-500		427.0	-213,500		60.0	-30,000	
Add:									
Nose Gear Up		185		243.0	44,955		91.0	16,835	
Main Gear Up		500		418.0	209,000		105.0	52,500	
Alighting Gear (Up)			792	376.5		298,174	95.7		75,755
Alighting Gear Retraction						+310			+28,235

WEIGHT AND BALANCE

Derivation of Direct Lift Mode
GROUP OR CONDITION

ENGINEER
CHECKED
DATE

ITEM	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ. DIST.	HORIZ. MOMENT 2	HORIZ. MOMENT 1	VERT. DIST.	VERT. MOMENT 2	VERT. MOMENT 1
Weight Empty (Composite Mode)			12,685	387.5		4,915,408	109.5		1,389,487
Removals			-865	443.7		-383,831	113.5		-98,191
Firewalls & Mounts (Lift Engine)		- 27		446	-12,042		115	- 3,105	
Firewalls & Mounts (Div. Pipes)		- 17		443	- 7,531		100	- 1,700	
Lift Engine (1)		-420		446	-187,320		115	-48,300	
Air Induction (Lift Engine)		- 7		446	- 3,122		138	- 966	
Starting (Lift Engine)		- 4		446	- 1,784		115	- 460	
Controls (Lift Engine)		- 7		300	- 2,100		115	- 805	
Ejector Cooling (Lift Engine)		- 2		446	- 892		100	- 200	
Diverter Valves (2)		-150		443	-66,450		125	-18,750	
Diverter Tail Pipes (2)		- 80		443	-35,440		125	-10,000	
Hover Control Piping		- 53		446	-23,638		83	- 4,399	
Doors (Exhaust & Air Induction)		- 98		444	-43,512		97	- 9,506	
Additions			1,061	443.8		470,871	114.6		121,611
Firewalls & Mounts (Lift Engines)		54		446	24,084		115	6,210	
Lift Engines (2)		840		446	374,640		115	96,600	
Air Induction (2)		14		446	6,244		138	1,932	
Starting		8		446	3,568		115	920	
Controls		14		300	4,200		115	1,610	
Ejector Cooling		4		446	1,784		100	400	
Lift/Cruise Tail Pipes (2)		45		443	19,935		125	5,625	
Hover Control Instal.		30		446	13,380		83	2,490	
Doors (Exhaust & Air Ind.)		52		443	23,036		112	5,824	
Weight Empty (Direct Lift Mode) (Gear Down)			12,881	388.4		5,002,448	109.7		1,412,907
Weight Empty Change			+196			+87,040			+23,420
Useful Load Change (016)			+ 7	446		+ 3,122	83		+ 581
Operating Weight Empty Change			+203	444.1		90,162	118.2		24,001

NORTHROP CORPORATION
NORAIR DIVISION

WEIGHT AND BALANCE

Direct Lift Mode
GROUP OR CONDITION

ENGINEER
CHECKED
DATE

MODEL N-309
REPORT NO.

ITEM	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ. DIST.	HORIZ. MOMENT 2	HORIZ. MOMENT 1	VERT. DIST.	VERT. MOMENT 2	VERT. MOMENT 1
Composite Mode Operating Wt. Empty (Gr.Dn)			13,193	382.7		5,049,487	110.1		1,452,343
Operating Weight Empty Change			203	444.1		90,162	118.2		24,001
Direct Lift Operating Weight Empty (Gr.Dn)			13,396	383.7		5,139,649	110.2		1,476,344
Variable Stability System			500	354.5		177,227	101.2		50,591
Gross Weight Less Fuel (Gear Down)			13,896	382.6 10.8% MAC		5,316,876	109.9		1,526,935
Fuel			2,404	397.0		954,298	104.3		250,758
Fwd. Fuel		1,110		292.0	324,120		107.0	118,770	
Aft Fuel		1,294		487.0	630,178		102.0	131,988	
VTOL - Direct Lift (Gear Down)			16,300	384.7 13.5% MAC		6,271,174	109.1		1,777,693

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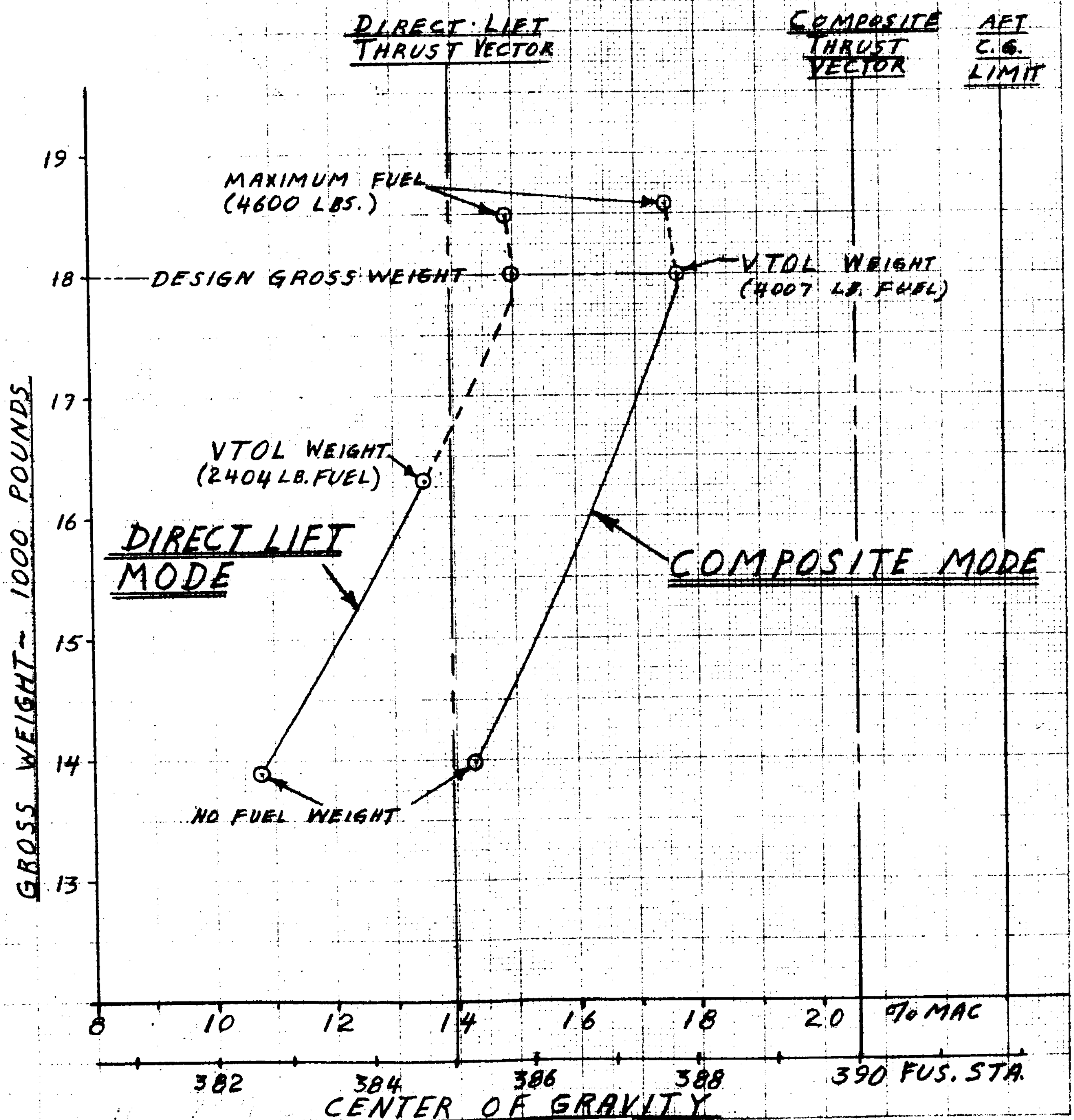
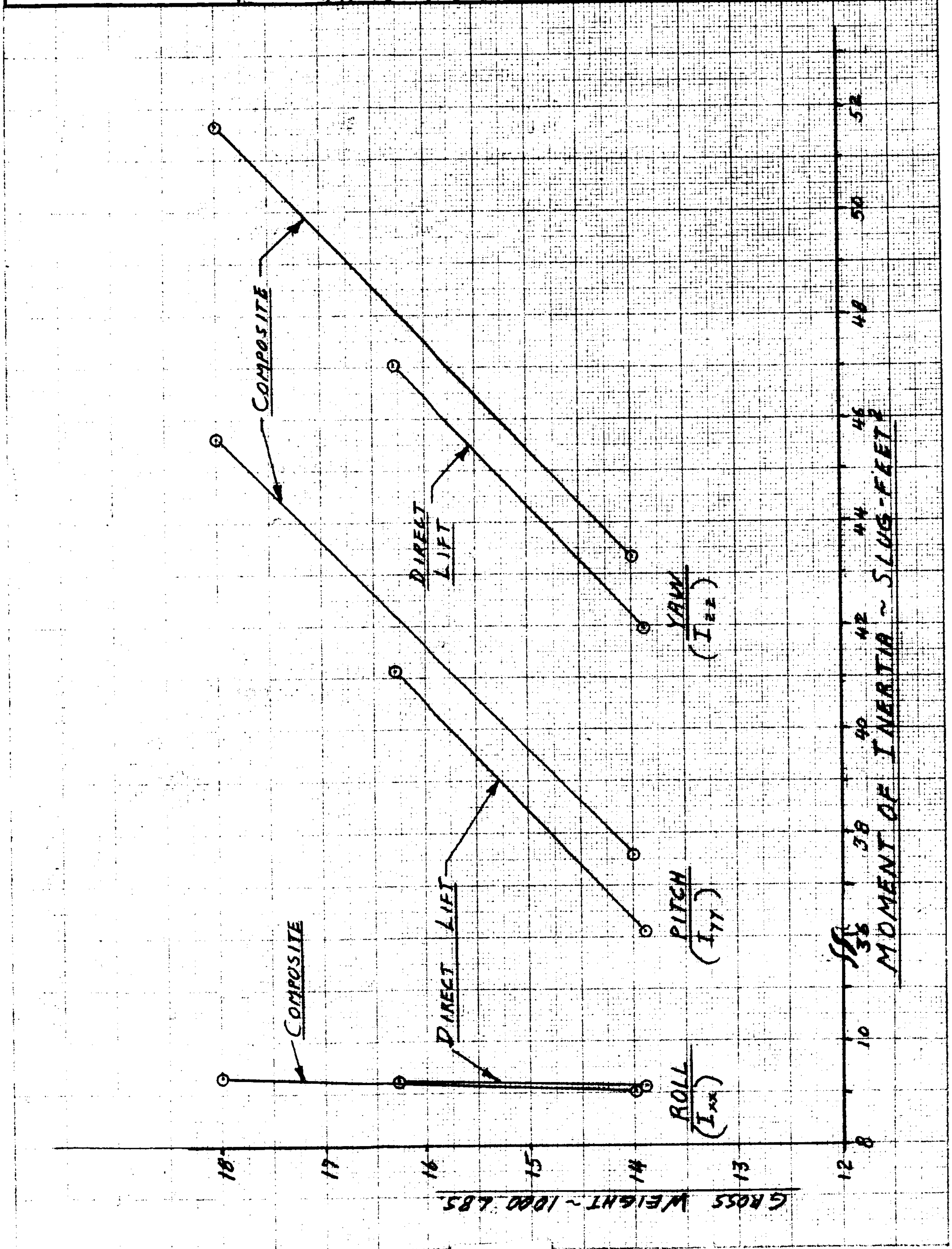
CHECKER		REPORT NO.
DATE	CENTER OF GRAVITY DIAGRAM	MODEL N-309

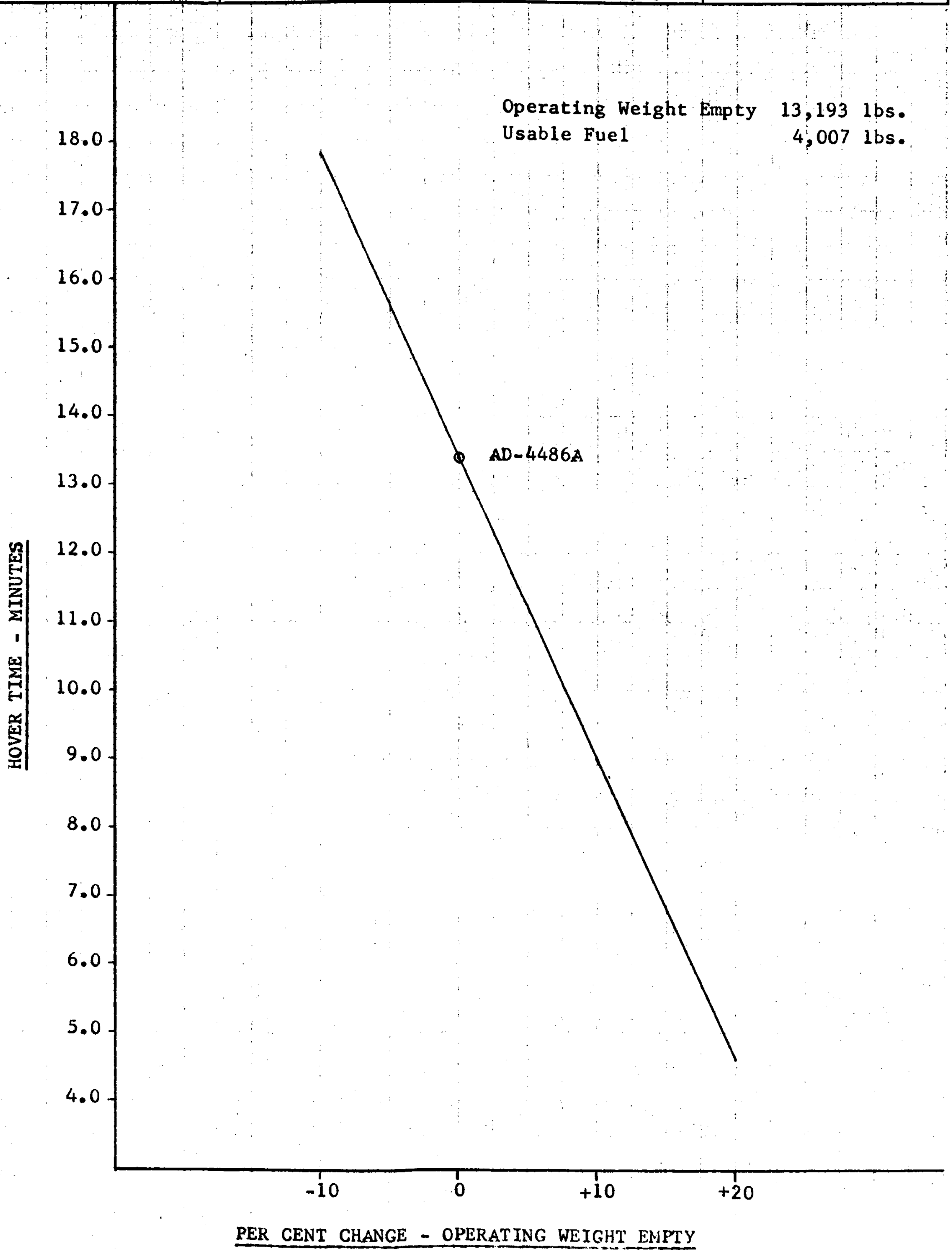
NOTES

1. GEAR DOWN CONDITION SHOWN. EFFECT OF GEAR RETRACTION IS NEGLIGIBLE.
2. TWO MAN CREW CONDITION SHOWN. EFFECT OF AFT MAN OUT IS APPROXIMATELY +1.8% (+1.4 IN.) AT 18,000 LBS. AND +2.4% (+1.8 IN.) AT 14,000 LBS.
3. AFT C.G. LIMIT AS SHOWN. THE FORWARD C.G. LIMIT TO BE DETERMINED BY STRUCTURAL LIMITATION.



ENGINEER	NORTHROP CORPORATION NORAIR DIVISION	PAGE 50
CHECKER		REPORT NO.
DATE	MOMENT OF INERTIA VERSUS GROSS WEIGHT	MODEL N-309





Operating Weight Empty 13,193 lbs.
Usable Fuel 4,007 lbs.
AD-4486A
HOVER TIME - MINUTES
18.0
17.0
16.0
15.0
14.0
13.0
12.0
11.0
10.0
9.0
8.0
7.0
6.0
5.0
4.0
-10
0
+10
+20
PER CENT CHANGE - OPERATING WEIGHT EMPTY

The weight estimate presented for the Northrop Norair version of the NASA V/STOL Research Airplane Design Study is based on empirical formulae, calculated and actual weight estimates.

The supporting data included in this section cover the calculations, formulae, basic curves, allowances for special features, and assumptions and other means for arriving at the estimated weights of the structural, power plant and equipment groups.

Structural weights are based on the following design criteria referenced in the design criteria report.

Basic Flight Design Gross Weight = 18,000 lbs.

Flight Load Factor - 3.75 limit

Limit Sea Level Speed - 400 K

Basic Mission VTOL Gross Weight - Composite 18,000 lbs.

Basic Mission VTOL Gross Weight - Direct Lift 16,300 lbs.

Equipment weights are predicated on the design requirements furnished by the contracting agency. Instrumentation and furnishing weights are consistent with those required for a two man crew to effectively fulfill the design mission.

Wing Group

The wing group weight estimate is estimated by using the following empirical formula:

$$W_w = .008305 \left[(Wg\eta)^{.620} (ARst)^{.590} (S)^{.640} \left(\frac{1+\lambda}{2TR}\right)^{.36} \left(1t \frac{V}{500}\right)^{.500} \right]^{.914}$$

where,

W_w	=	Basic Flight Design Gross Weight	=	18,000 lbs.
η	=	Ultimate Flight Load Factor	=	5.625
ARst	=	Structural Aspect Ratio = AR/CoSc/4	=	6.8
S	=	Planform Wing Area	=	210 sq. ft.
λ	=	Planform Taper Ratio - Ct/Cr	=	.400
T.R	=	Thickness Ratio - Root Chord	=	13%
V	=	Sea Level Limit Speed	=	400 knots
Ω_{c4}	=	Sweep Angle - 1/4 Chord	=	20°

The estimated weight for the wing utilizing this basic formula is 833 lbs. An additional weight is calculated for the leading edge flap and back up structure of 97 lbs. resulting in a total wing weight of 930 lbs. The basic wing weight determined above is plotted on the wing estimation chart on page 54 for comparative purposes with other model airplanes.

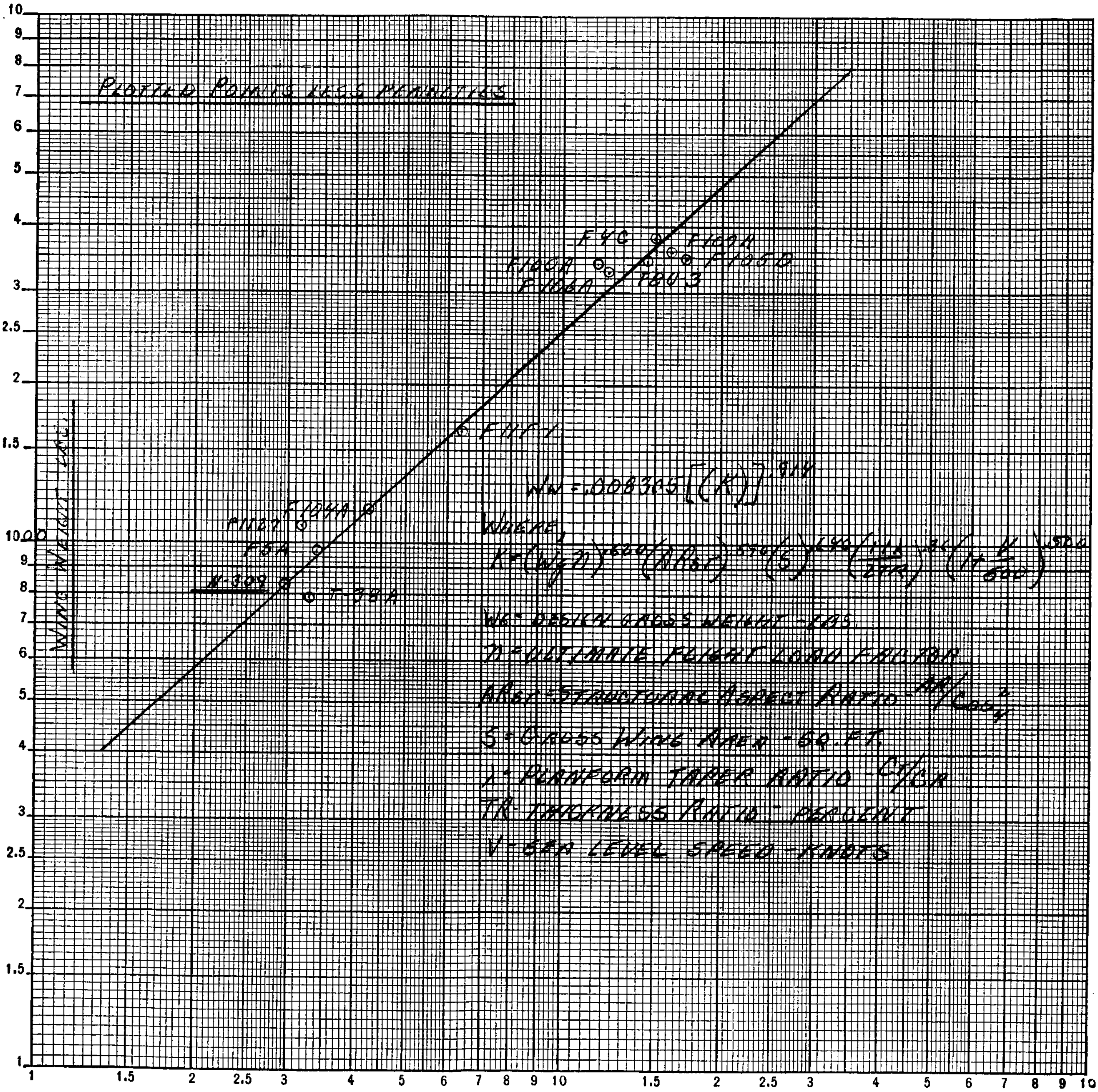
Total Wing Weight 930 lbs.

Tail Group

The weight of the tail group is estimated as a total group and also by considering the horizontal and vertical components separately and applying Northrop Norair developed formulae.

WING WEIGHT ESTIMATION

FIGHTERS & INTERCEPTORS



$$K \times 10^{-5} = (W_G N)^{.620} (AR_{ST})^{.590} (S)^{.640} \left(\frac{H\lambda}{2TR}\right)^{.5} \left(1 + \frac{V}{500}\right)^{.500}$$

The tail group weight estimate as a total group is a function of the wing weight in the following formula:

$$W_t = 1.072 \left[W_w \right]^{.842}$$

where,

W_t = Total Tail Group Weight - lbs.

W_w = Estimated Basic Wing Weight - 833 lbs. (less penalties)

The use of this formula results in an estimated tail group weight of 309 lbs.

In order to further substantiate the individual horizontal, vertical and ventral component weights the following formulae were utilized:

Horizontal Tail

$$W_h = 16.55 \left[\frac{W_g \eta S_r \; bs/t \; \bar{C}w/Lt}{10^6} \right]^{.4606}$$

where,

W_h	=	Horizontal Tail Weight - lbs.		
W_g	=	Basic Flight Design Gross Weight	=	18,000 lbs.
η	=	Ultimate Flight Load Factor	=	5.625
S_t	=	Theoretical Tail Area	=	67.4 sq. ft.
bs	=	Structural Span = Span/Bos 1/4	=	237.3 ins.
t	=	Thickness Theoretical Root Chord	=	7.9 ins.
$\bar{C}w$	=	Wing M.A.C.	=	75.3 ins.
Lt	=	Tail Length - 1/4 chord wing mac to 1/4 chord horiz. tail mac	=	180.7 ins.

The above parameters used in the basic formula results in a horizontal tail weight estimate of 128 lbs.

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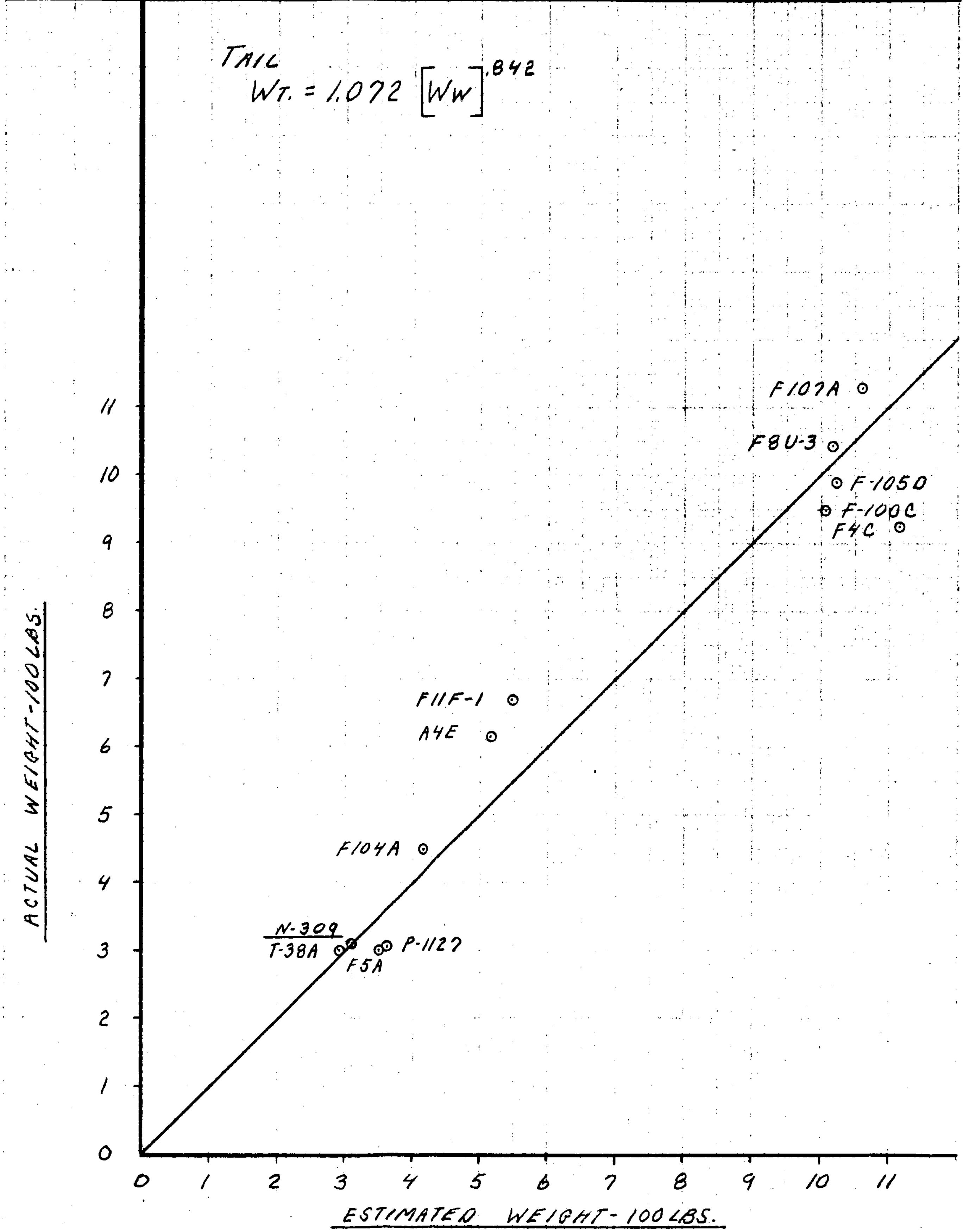
DATE	TAIL WEIGHT ESTIMATION	MODEL



Vertical Tail

$$W_v = .02184 \left[(Wg\eta)^{.432} \; (S_v)^{.791} \; (bs/hr)^{.464} \right]^{.952}$$

where,

W_v = Vertical Tail Weight - lbs.

Wg = Basic Flight Design Gross Weight = 18,000 lbs.

η = Ultimate Flight Load Factor = 5.625

S_v = Vertical Tail Area = 48 sq. ft.

bs = Structural Span = Span/cos 1/4 = 155.3 in.

hr = Max. Thickness Root Chord = 11.7 in.

The use of these design parameters results in a vertical tail weight of 144 lbs.

In addition to these estimated tail group weights, the dorsal fairing is estimated at 2.0 lbs/sq./foot resulting in an additional weight of 7 lbs., and the ventral fin/skid is estimated at 4.0 lbs. a square foot for 28 lbs.

Summating the individual component weight estimates,

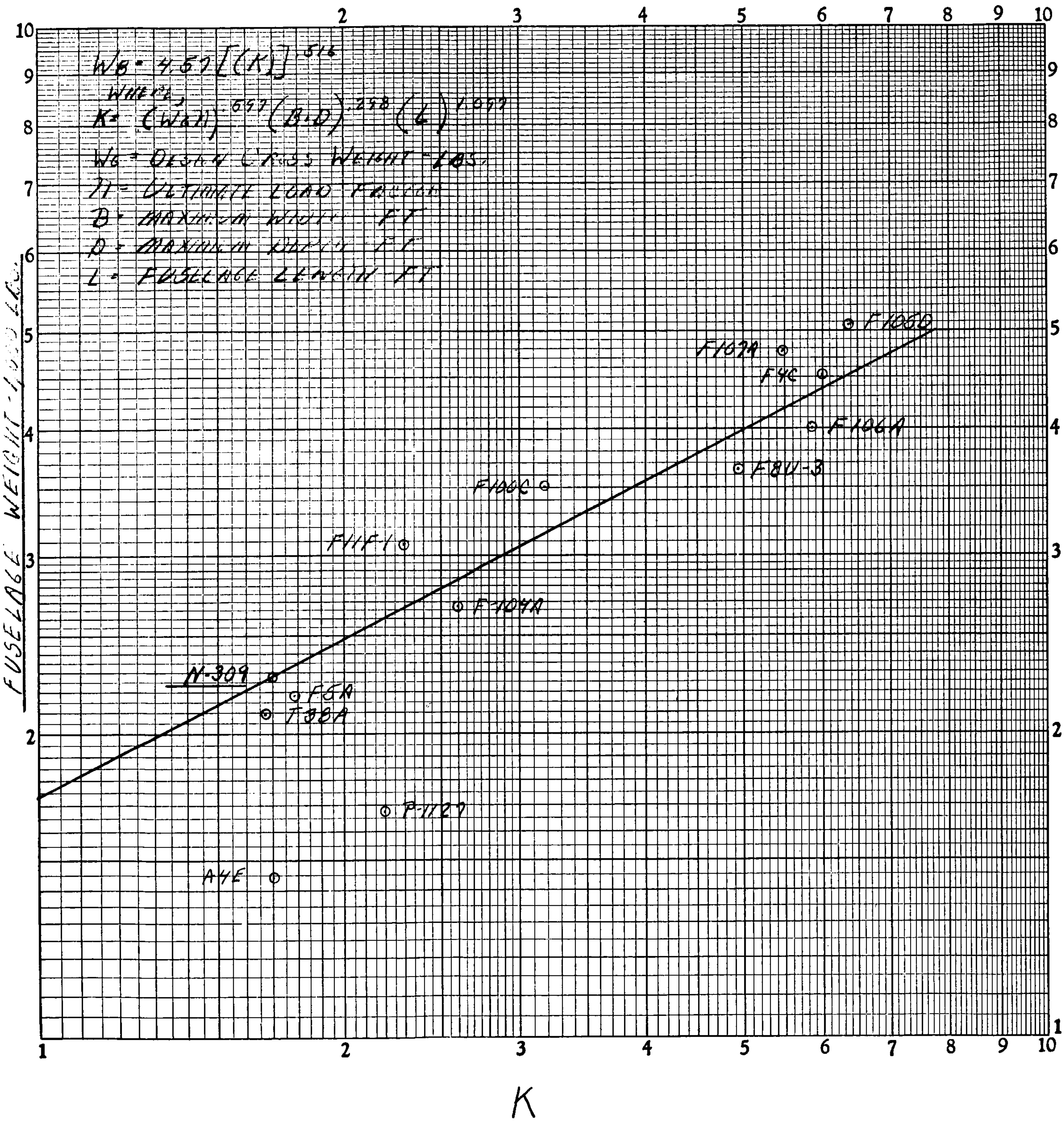
		Weight - Lbs.
Horizontal Tail Weight	=	128
Vertical Tail Weight	=	144
Dorsal Fairing Weight	=	7
Ventral Fin/Skid Weight	=	28

Component Tail Group Weight = 307 lbs.

This component weight breakdown of 307 lbs. agrees favorably with the weight estimate of the total tail group weight of 309 lbs. which is used as the tail group weight estimate. A plot of the total tail group weight estimate is included on page 56.

Tail Group Weight 309 lbs.

Body Weight Estimation
Fighters & Interceptors



Body Group

The estimated weight of the body group is calculated by using the following formula depicted in chart form for comparative purposes on page 58.

$$W_b = 4.57 \left[(Wg\eta)^{.597} \; (B \cdot D)^{.298} \; (L)^{1.097} \right]^{.516}$$

where,

W_b	=	Basic Body Weight - lbs.		
Wg	=	Basic Flight Design Gross Weight	=	18,000 lbs.
η	=	Ultimate Flight Load Factor	=	5.625
B	=	Max. Body Width	=	5.667 ft.
D	=	Max. Body Depth	=	5.25 ft.
L	=	Structural Body Length	=	43.9 ft.

The estimated basic body weight using the above parameters is 2,280 lbs. A comparative plot is presented on page 58.

A correlation plot of the total wing and body weight estimates of the comparative model airplanes is depicted on page 60.

A weight penalty is assessed to the basic body weight for special design features required for a VTOL airplane, i.e. lift engine air induction and exhaust doors as follows:

Air Induction Doors - Lift Engine 2.0 lbs/sq/ft	76 lbs.
Exhaust Doors - Lift Engine 2.0 lbs/sq/ft	60 lbs.
Exhaust Doors - Lift/Cruise Engine 8.0 lbs/sq/ft	42 lbs.
Mechanism - Lift Engine Doors - 7.0 lbs. engine	49 lbs.
Mechanism - Lift/Cruise Doors - 15.0 lbs. engine	30 lbs.
Total Estimated Weight for Special Features	257 lbs.
Basic Estimated Body Weight	2,280 lbs.
Body Group Weight	2,537 lbs.

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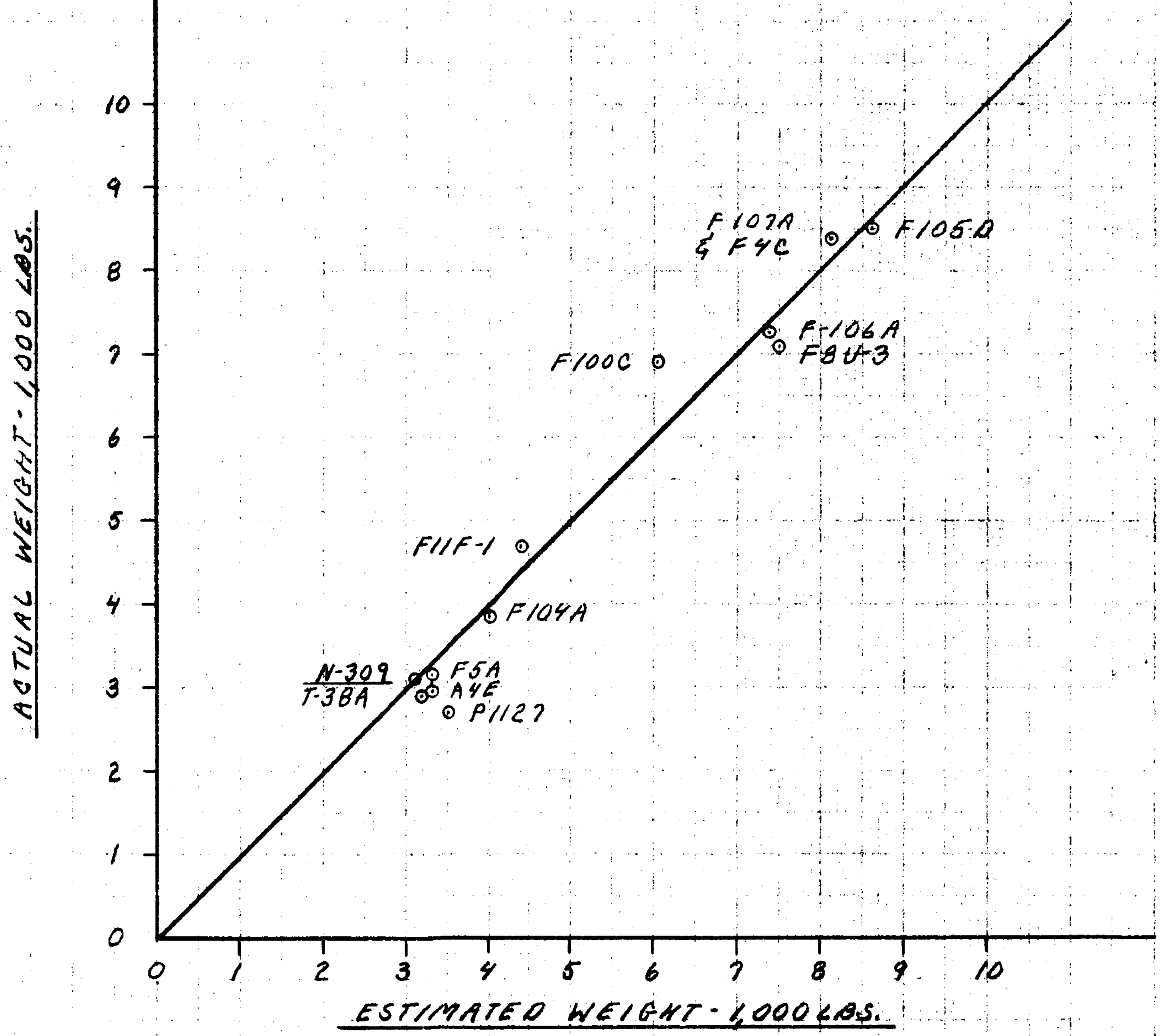
CHECKER	CORRELATION	REPORT NO.
DATE	WING & BODY WEIGHTS	MODEL

WING

$$W_W = .008305\left[(W_G N)^{.620}(AR_{BT})^{.590}(S)^{.690}\left(\frac{1+\lambda}{2 T A}\right)^{.36}\left(1+\frac{V}{500}\right)^{.50}\right]^{.914}$$

BODY

$$W_B = 4.57\left[(W_G N)^{.597}(B \cdot D)^{.298}(L)^{1.097}\right]^{.616}$$



Alighting Gear

The alighting gear group weight is normally estimated using design gross weight and load factor as given by the equation in the chart on page 62. In this particular airplane, however, the cost factors involved in designing a 100 percent new alighting gear dictated the utilization of existing components as far as possible, and results in a somewhat less than optimum weight gear.

The base alighting gear weight estimate is calculated from the following formula for the accuracy analysis statement.

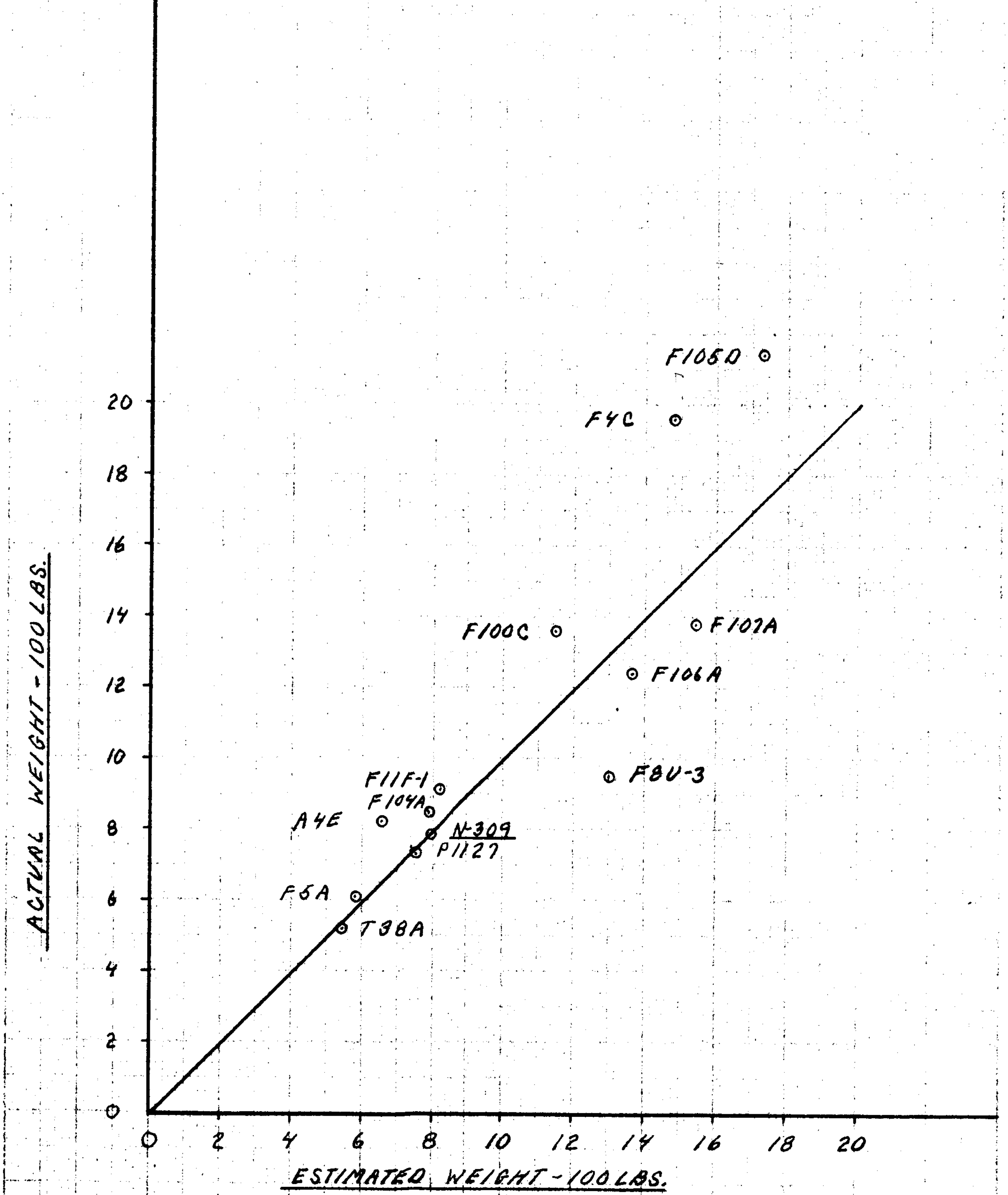
$$Wg = .05195 \left[Wg\eta \right]^{.801}$$

The use of this equation results in an alighting gear weight estimate of 531 lbs.

In order to further substantiate the estimated weight of the alighting gear the following component weights are presented.

Nose Gear A4E - Report 31388					192 lbs
Rolling Stock				26 lbs	
Struct				159 lbs	
Controls				7 lbs	
Main Gear					600 lbs
Rolling Stock				170 lbs	
Wheels	F-5A		46		
Tires	F-5A	Actual	49		
Brakes	F-5A	Modified	75		

$$W_G = .06195 \left[W_G N \right]^{.801}$$



The main gear strut weight is based on the following formula,

$$W_s = .001514 \left[(W_l)^{.776} \quad (\eta_l)^{.167} \quad (L)^{.768} \right] 1.11541$$

where,

W_s = Total Strut and Brace Weight

W_l = Landing Weight = 18,000 lbs.

η_l = Landing Load Factor = 2.5

L = Struct Length - ℄ Trunnion to ℄ axle - fully extended = 68 ins.

The calculated strut weight based on the above parameters is	322 lbs.
Shrink Mechanism	8 lbs.
Controls Weight - similar to the F-5A	100 lbs.
Alighting Gear Group Weight used for estimate	792 lbs.

Surface Controls, Hydraulics & Electrical Groups

The weight of the surface controls, hydraulic and electrical groups are estimated as a single group by the following empirical equation.

$$W_{she} = 2.12 \left[(W_g \eta)^{.623} \quad (L+B)^{.224} \quad (V_{sl})^{.167} \right]^{.654}$$

where,

W_{she} = Total Weight of Surface Control, Hydraulic & Electrical Groups (less special features)

W_g = Basic Flight Design Gross Weight = 18,000 lbs.

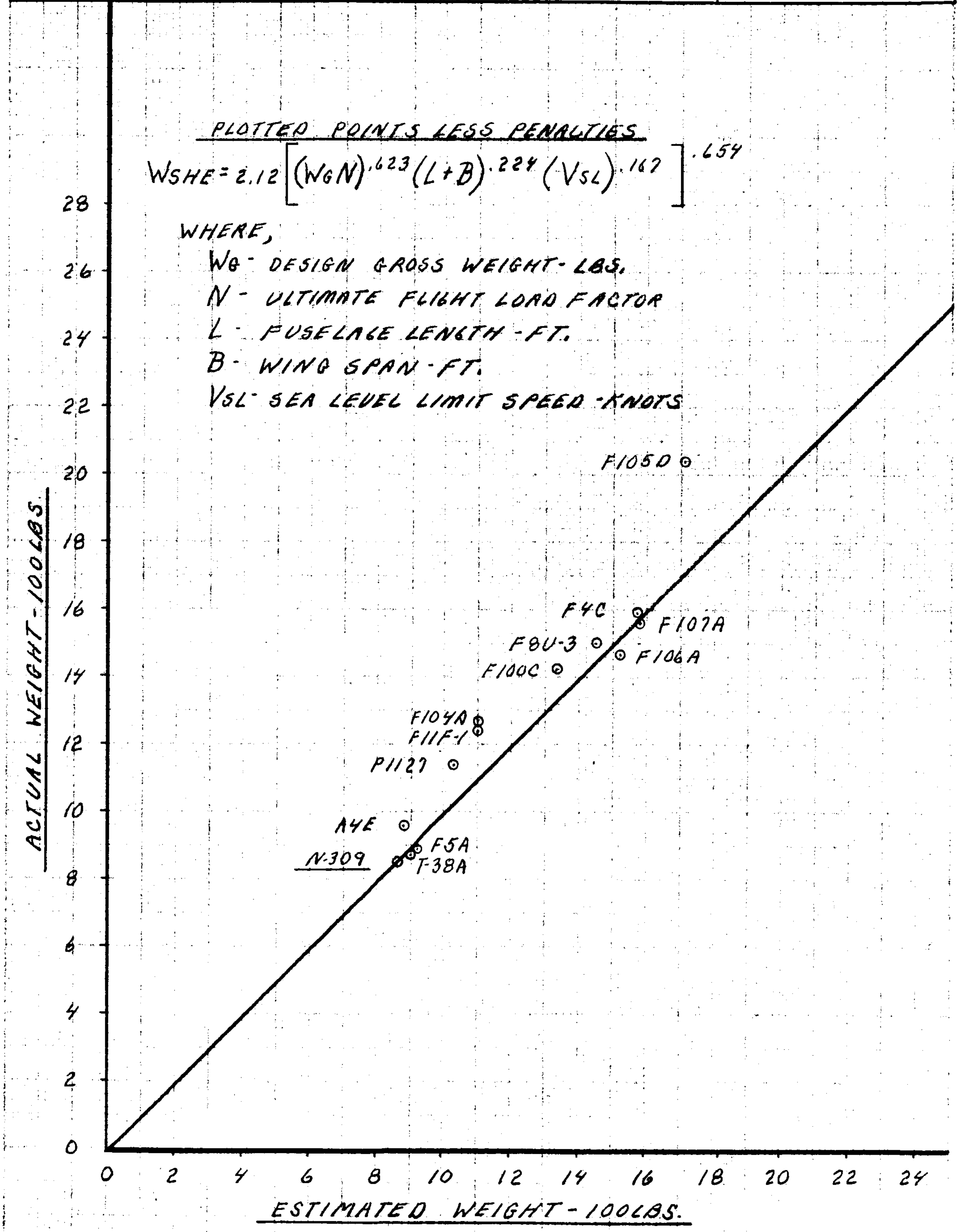
η = Ultimate Flight Load Factor = 5.625

L = Fuselage Length = 45.5 ft.

B = Wing Span = 35.5 ft.

V_{sl} = Sea Level Limit Speed = 400 knots

ENGINEER	NORTHROP CORPORATION NORAIR DIVISION	PAGE 64
CHECKER	CORRELATION	REPORT NO.
DATE	SURFACE CONTROLS, HYDRAULICS & ELECTRICAL GROUP WEIGHTS	MODEL



The weight estimate derived from the above parameters for these design groups is 850 lbs. A plot of this formula is included on page 64 for comparison with other model aircraft.

In order to check this empirical weight estimate and to provide adequate weight distribution for balance computations, the individual systems are analyzed from preliminary system drawings and are entered in detail on the Detail Weight Statement, AN-9102D. The resulting group weights are as follows:

	Weight-Lbs
Surface Controls Group	508
Hydraulic Group	190
Electrical Group	335

The summation of the above individual group weight estimates is 1,033 lbs. or 183 lbs. over the empirical weight analysis. This additional weight increment is attributable to the surface control group special features, stability augmenter system, leading edge flap controls, and reaction nozzle boost actuators.

Total Surface Controls, Hydraulic & Electrical Groups 1,033 lbs.

Engine Section

The engine section weight estimate is obtained by calculating the components weights from the lift/cruise and lift engine layouts.

Lift/Cruise Engine Bay		491 lbs.
Nacelles		
Structure 2.0 lbs. sq. ft. - 74.8 sq. ft. ea.	291	
Blast Plates .10 steel - 951 sq. in. ea.	54	

Mounts			
Engine & Diverter Valve - Calculated		22	
Tail Pipe & Nozzles - Calculated		26	
Firewalls & Blankets			
Firewalls - .016 steel + 5% attach	9,930 sq. in.	95	
Blankets - Minimum Requirement		3	
Lift Engine Bays - 7 engines			189 lbs.
Mounts			
Structure Mounted Mounts	4.0 lbs. eng.	28	
Trunnions	3.0 lbs. eng.	21	
Engine Mounted Mounts	2.0 lbs. eng.	14	
Firewalls - .016 Steel	18.0 lbs. eng.	126	
	Total Engine Section		680 lbs.

Propulsion Group

The weight of the Propulsion Group is estimated by components as shown below.

Lift/Cruise Engine Installation

Engine Installation

The engine weight for the General Electric YJ-85 GE-19 as quoted in Model Specification E1129 is as follows:

Engines (2) YJ-85 GE-19 Dry Weight	764	
Bleed Tubes (2)	4	
Insulation (2)	6	
Residual Fluids (2)	10	
Total Engine Installation		784 lbs.

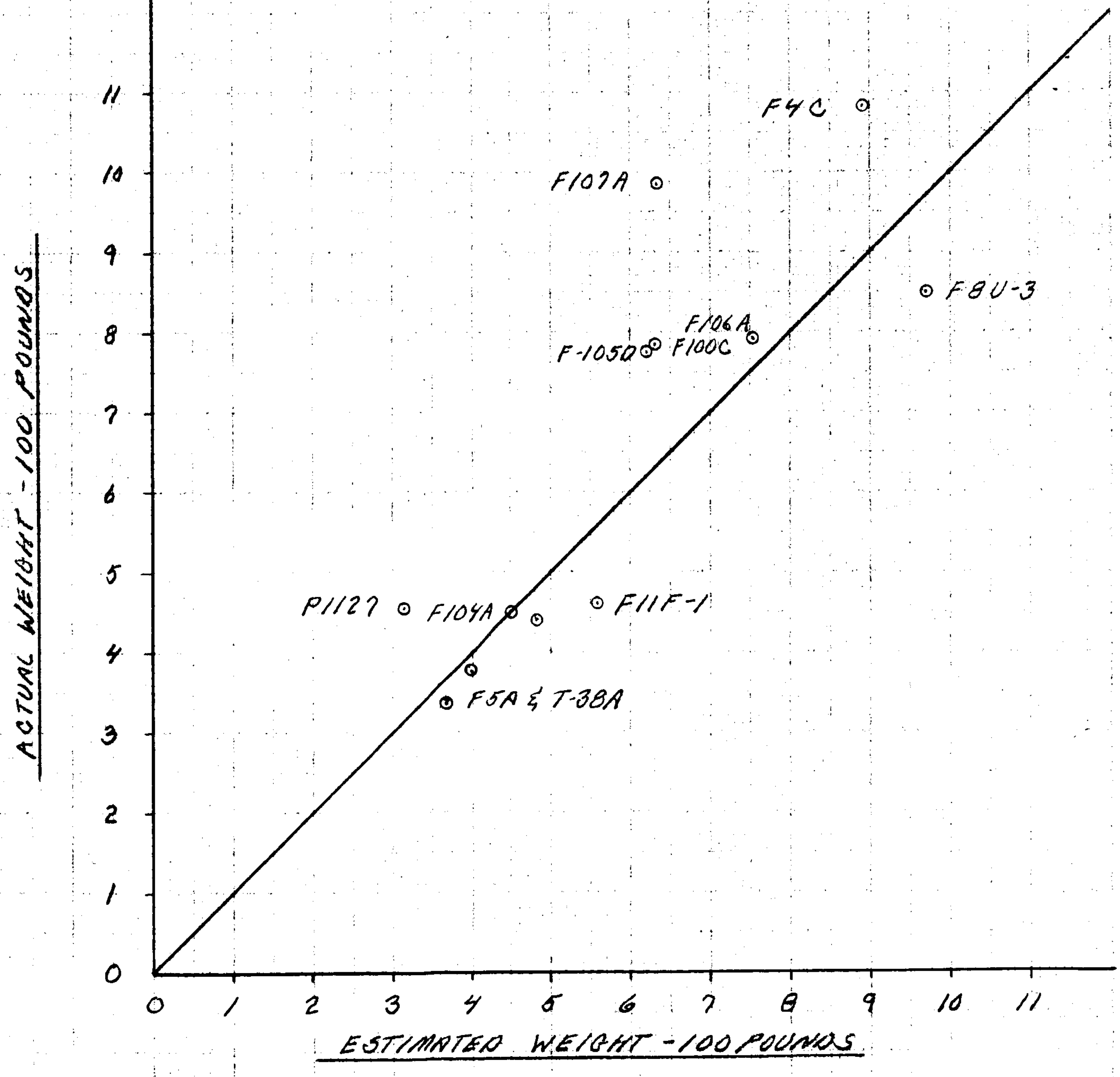
Diverter Valves

The diverter valve weight is referenced in Vol. II - G.E. Diverter Valve Final Report.

Diverter Valve (2)		150 lbs.
Diverter Tail Pipe		
Pipe .050 Steel (2)	50	
Bellows - 2 Req. Per Engine	20	
Flanges & Bolts 5.0 lbs. Engine	10	
Total Diverter Tail Pipe		80 lbs.
Cruise Tail Pipe - .050 Steel		44 lbs.
Ejectors - Allowance 7.0 lbs. Engine		14 lbs.
Accessory Drives - CSD - CF5A 70 lbs. ea.		140 lbs.
Air Induction - 2.0 lbs. sq. ft.		36 lbs.
Engine Controls		
Throttle Quandrants 2 - 7.5 lbs. ea.	15	
Amplifiers 2 - 1.5 lbs. ea.	3	
Actuators 2 - 3.0 lbs. ea.	6	
Wiring - Allowance	4	
Total Engine Controls		28 lbs.
Starting System - Air Impingement		
Impingement Fitting (L.H. only)	4	
Shutoff Valves	6	
Ducting & Bellows	5	
Starting System Total		15 lbs.

FUEL SYSTEM

$$W_{FS} = .696 \left[W_F \right]^{.760}$$



<u>Lube System</u> - Filler Necks, etc. 2 lbs.

<u>Fuel System</u>

The fuel system weight is estimated using the following formula for the accuracy analysis back up data.

$$W_{fs} = .695 \quad W_f^{\ .760}$$

where,

W_{fs} = Fuel System Weight

W_f = Total Fuel (including warm up fuel)

This calculation results in a fuel system weight of 400 lbs. In order to check this empirical method the fuel system was analyzed from layouts and the breakdown by components is shown in the detail weight statement, AN-9103D. The fuel system build-up by components results in a weight of 375 lbs., a 25 lb. reduction from the empirical method and is justified by the fact that there is no single point fueling and no wing fuel provisions.

Fuel System Weight	375 lbs.
Total Lift/Cruise Installation	1,668 lbs.

Lift Engine Installation - 7 Engines

Engine Installation

The engine weight for the General Electric YJ-85-GE-19 engine quoted in Model Spec E1129 is reduced by 5 lbs. per engine by the removal of the overspeed provisions.

Engines (7) YJ-85-GE-19 377 lbs. ea.	2,639	
Bleed Tubes (7) 2 lbs. ea.	14	
Insulation (7) 3 lbs. ea.	21	
Residual Fluids (7) 5 lbs. ea.	35	
Vector Nozzle (7) 33 lbs. ea.	231	
Lift Engine Ejectors	14	
Total Engine Installation		2,954 lbs.
Air Induction (7) 5.0 lbs. engine		35 lbs.
Starting System		
Ducting (7) 1.0 lbs. engine	7	
Starting Valves (7) 2.7 lbs. engine	19	
Total Starting System		26 lbs.
Controls		
Throttle Quadrants (inl. in lift-cruise)		
Amplifiers (7) 1.5 lbs. engine	10.5	
Actuators (7) 3.0 lbs. engine	21.0	
Installation	18.5	
Total Controls		50 lbs.
Lube System (7) 1.0 lb. engine		7 lbs.
Total Lift Engine Installation		3,072 lbs.
Total Propulsion Group		4,740 lbs.

Instrument Group

The instrument group weight is derived by applying actual and specification weights to the flight and engine instruments required for the fulfillment of the design mission. For each instrument, a weight allowance for installation of wiring, tubing and hardware is included. These weights are shown in detail in the Detail Weight Statement, AN-9102D.

Instrument Group Weight 171 lbs.

Hydraulic & Electrical Groups

The weights of these groups are included in the surface controls, hydraulic and electrical group weight derivation. A component breakdown based on the individual systems is detailed in the Detail Weight Statement.

Electronics Group

The electronics weight estimate is predicated on the navigation and communication requirements specified in the NASA Request for Proposal L-7151 Statement of Work. A compilation of this equipment by set designation is included in the Detail Weight Statement.

Electronics Group Weight 220 lbs.

Furnishings and Equipment Group

The Furnishings and Equipment Group weight is a composite of a number of individual estimated weights. These individual estimates are based on the use of existing equipment, where possible, vendor information, design drawing and sketches and weights of similar components and sub-systems used in other single and dual seat airplanes.

Accommodations For Personnel (305 lbs.)

Crew Seats

Item	Basis For Estimate	Unit Wt.	No. Req.	Total Wt.
Basic Seat	North American LW-2D	69	2	138
Lap Belt	North American LW-2D	2	2	4
Harness	North American LW-2D	3	2	6
Adjust. Mech.	Northrop T-38A	7	2	14
Catapult	North American LW-2D	24	2	48
Tracks & Supts.	Northrop T-38A	15	2	30
Cushion, etc.	North American LW-2D	20	2	40
Total Crew Seats				280

Oxygen System (1800 psi gaseous)

Item	Basis For Estimate	Unit Wt.	No. Req.	Total Wt.
Bottle	Zep Aero Co.	12	1	12
Regulators, etc.	Zep Aero Co.	2.5	2	5
Plumbing, est.	Northrop T-38A	--	-	8
Total Oxygen System				25

Miscellaneous Equipment & Furnishings (70 lbs.)

Miscellaneous Equipment

Item	Basis For Estimate	Unit Wt.	No. Req.	Total Wt.
R. V. Mirror	T-38A	1	1	1
Glare Shields	T-38A	2	2	4
Inst. Panels	T-38A	2	8	16
Consoles, etc.	T-38A	-	-	23
Total Miscellaneous Equipment				44

Furnishings

Item	Basis For Estimate	Unit Wt.	No. Req.	Total Wt.
Data Cases	T-38A	--	--	4
Soundproofing	Allowance	--	--	17
Misc. Stowage	Allowance	--	--	5
Total Furnishings				26

Emergency Equipment (35 lbs.)

Item	Basis For Estimate	Unit Wt.	No. Req.	Total Wt.
Control Unit	T-38A (J85 engines)	1.50	1	1.5
Detectors	T-38A (J85 engines)	1.65	10	16.5
Control Box	T-38A	1.00	10	10.0
Wiring	Allowance	--	--	7.0
Total Emergency Equipment				35.0

Total Furnishings and Equipment Group 410 pounds

Air Conditioning and Anti-Icing Group (68 lbs.)

The air conditioning and anti-icing system is a combined heating and cooling system which utilizes engine bleed air as the heat source. The air is extracted from both lift/cruise engines final compressor stage and passed through ducts to the heat exchanger and expansion turbine and then on to the cockpits and equipment bays. The air is exhausted from the cockpits to the equipment bays for component cooling. The aft bay cooling will be fan augmented. Since cabin pressurization is not required, the system is not sealed beyond that necessary to satisfy the air conditioning requirements. No penalty was assessed for pressurization equipment.

Air Conditioning

The weight of this system was obtained by analysis of preliminary schematic diagrams, by using data from the T-37A trainer, the Northrop T-38A, and from information supplied by The Garret Corporation concerning the weight of the heat exchanger/turbine package. The following is a breakdown of the estimated weights of this system:

Item	Source of Information	Weight, Pounds
Heat Exchanger & Turbine	The Garret Corp.	25
Water Separator	Cessna T-37A	2
Fans & Blowers	Northrop T-38A	4
Ram Air Valves	Northrop T-38A	4
Cockpit Controls	Cessna T-37A	7
Ducting & Scoops	Weight Allowance	19
Total Air Conditioning System		61

Anti-Icing

The only anti-icing system weight allowance made was for windshield and canopy de-fog system, and is broken down as follows:

Item	Source of Information	Weight, Pounds
Ducting	Preliminary Schematics	5
Valves	Cessna T-37A	1
Cockpit Controls	Cessna T-37A	1
Total Anti-Icing		7

Hover Controls Group

The Hover Controls Group is defined as the pneumatic portion of the hovering controls and includes the weight of the ducting, insulation and reaction control valves. The weight of the electronic controls, electro-hydraulic servos and actuation mechanisms is included in the Surface Controls Group and in the Variable Stability System.

The ducts are assumed to be connected by means of bolted flanges and/or clamped joints, with load relieving bellows welded to the ducts and appropriately spaced throughout the system.

The bellows and bolted flange weights are estimated from the graph on page 77. The equations of these curves were derived from weight calculations based on the figures shown above this graph. Bolted flanges are used to join all ducts with diameters larger than 4 inches and on the small ducts that attach to the engine bleed ports. Clamped couplings are used on the balance of the ducts. The clamped coupling weights used are the Marmon J-11 series clamped joints listed in the Marmon Company catalogue.

The insulation is used on the forward and aft fuselage ducts, and on the wing ducts from the fuselage mold line to wing station 135. Outboard of wing station 135, the wing is assumed to have titanium structure in the duct area. No insulation is used in the engine bay. The insulation weights are based on a 0.5 inch thick blanket with a density of 7 pounds per cubic foot with 2 mil steel face sheets on each side. The insulation weights were calculated to be .08951 pounds per linear inch for the 8.5 inch ducts and 0.04475 pounds per linear inch for the 4 inch diameter ducts.

The reaction control valve weights were calculated from preliminary design drawings.

The fairing covering the aft control valve is estimated to be 25 pounds.

The tip pod weight was calculated at 15 pounds each. The surface area of each tip pod was calculated to be 8.17 square feet. The weight per unit area is 1.84 pounds per square foot. This agrees closely with the value obtained for a smaller tip fairing used on a previous design (N-289).

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The following is the estimated hardware inventory for the Hover Control System.

Item	Location	Size (Stainless Steel)	Wt. Lbs.
Ducts			(147)
	Fwd. Fus. Duct	8.5 dia x 155 long x .020 wall	24
	Cntr. Fus. Manifold	8.5 dia x 115 long x .040 wall	35
	Aft Fus. Duct	8.5 dia x 160 long x .020 wall	24
	Cntr. Fus. Manifold	6.0 dia x 70 long x .036 wall	14
	Cntr. Fus. Wing T. O.	5.5 dia x 3 long x .020 wall	Neg.
	Cntr. Wing Ducts	4.0 dia x 50 long x .020 wall	4
	Outr. Wing Ducts	4.0 dia x 420 long x .020 wall	30
	Engine Bleed Ducts	3.0 dia x 96 long x .020 wall	5
	Engine Bleed Ducts	2.0 dia x 240 long x .020 wall	9
	Engine Bleed Ducts	1.5 dia x 60 long x .020 wall	2
Transition Ftgs. ("Y" and "T" Ftgs.)			(36)
	Cntr. Fus.	8.5 dia to 6.0 dia x .036 wall (2 req)	16
	Wing Take-off Tee	5.5 dia to 4.0 dia x .020 wall (1 req)	2
	Engine Bleed	3.0 dia to 2.0 dia x .020 wall (6 req)	7
	Engine Bleed	2.0 dia to 1.5 dia x .020 wall (14 req)	11
Bellows			(89)
	Fuselage Ducts	8.5 dia. 8 req @ 4.2 ea.	34
	Fuselage Ducts	6.4 dia. 4 req @ 2.3 ea.	9
	Wing Take-off Duct	5.5 dia. 1 req @ 1.7 ea.	2
	Wing Ducting	4.0 dia. 6 req @ 1.2 ea.	7
	Engine Bleed	3.0 dia. 12 req @ .9 ea.	11
	Engine Bleed	2.0 dia. 14 req @ .6 ea.	8
	Engine Bleed	1.5 dia. 37 req @ .5 ea.	18
Clamped Couplings (Marmon J-11 series including flanges & gaskets)			(22)
	Wing Ducts	4.0 dia. 2 req @ .970 ea.	2
	Eng. Bleed Ducts	3.0 dia. 6 req @ .785 ea.	5
	Eng. Bleed Ducts	2.0 dia. 16 req @ .498 ea.	8
	Eng. Bleed Ducts	1.5 dia. 20 req @ .333 ea.	7
Bolted Flanges			(50)
	Fuselage Ducts	8.5 dia. 12 req @ 2.7 ea.	32
	Wing Ducts	4.0 dia. 8 req @ 1.2 ea.	10
	Eng. Bleed Ducts	2x2 square 24 req @ .15 ea.	4
	Eng. Bleed Ducts	1.5x1.5 square 36 req @ .10 ea.	4
Insulation (.50 inch thick blanket with density = 7 lb/ft^3 and 2 mil steel inner and outer face sheets)			(35)
	Fwd. Fus. Ducts	8.5 dia. @ .08951 lb/in x 155 in.	14
	Aft Fus. Ducts	8.5 dia. @ .08951 lb/in x 160 in.	14
	Wing Ducts	4.0 dia. @ .04475 lb/in x 150 in.	7

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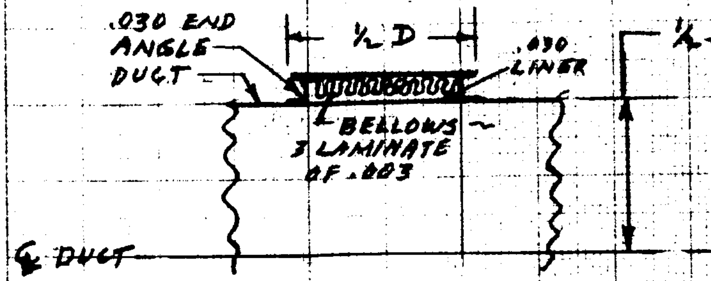
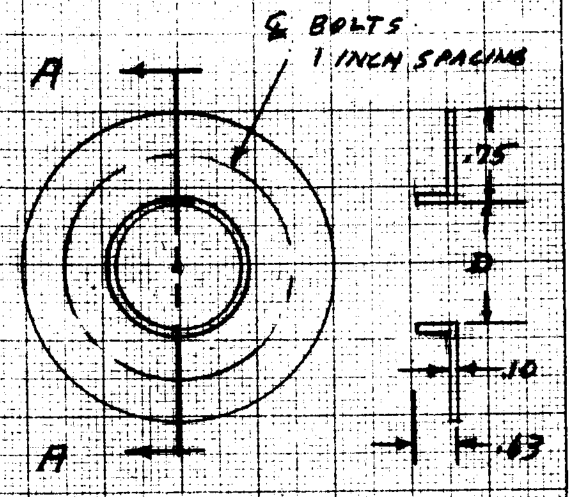


DIAGRAM OF A BELLOWS



D = TUBE DIAMETER

DIAGRAM OF A BOLTED FLANGE

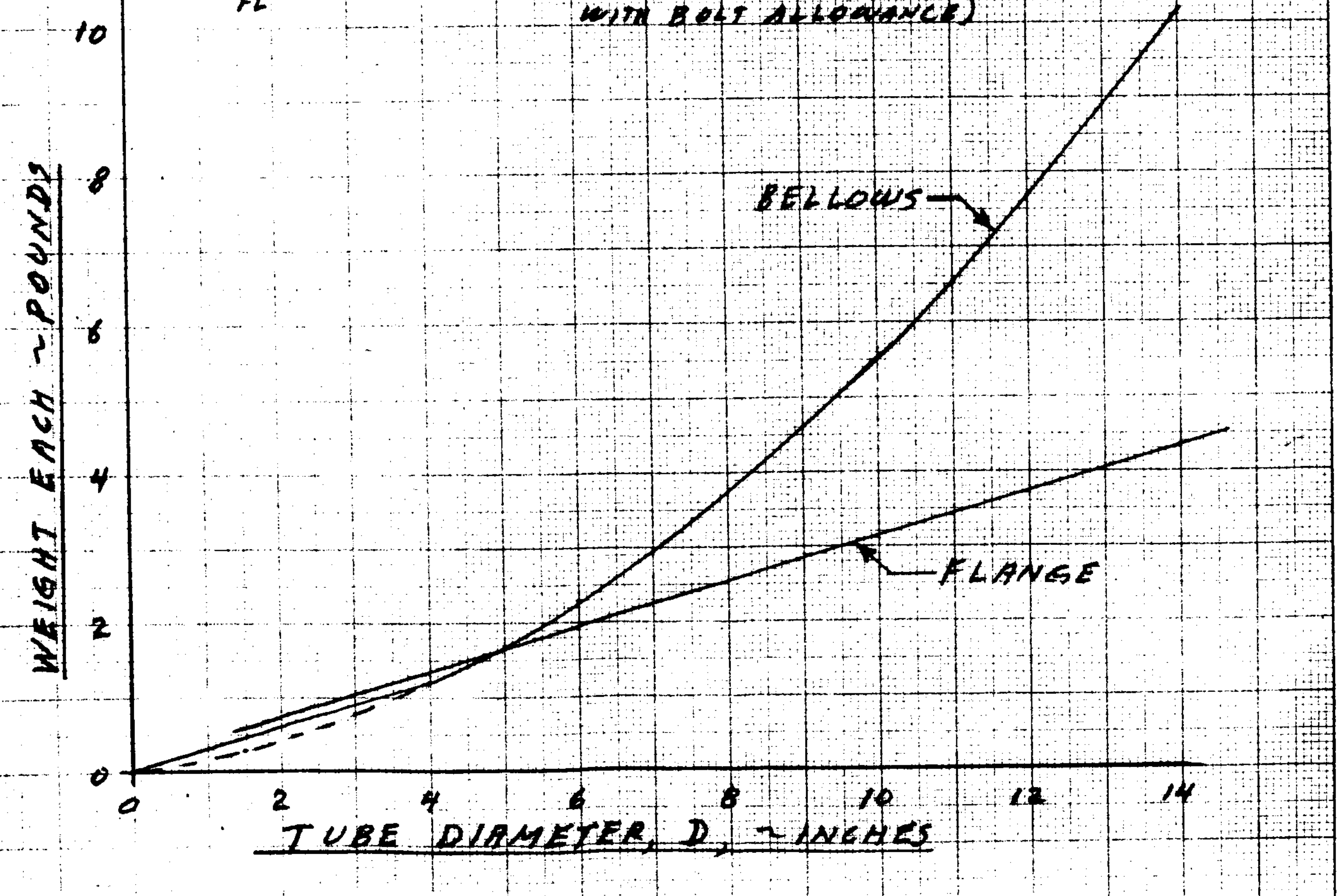
BELLOWS WEIGHT

$W_{BEL} = .04331\,D^2 + .12181\,D \quad (D \geq 4 \text{ INCHES})$

$W_{BEL} = .295\,D \quad (D \leq 4 \text{ INCHES})$

FLANGE WEIGHT

$W_{FL} = .303\,D + .130$ (SINGLE FLANGE WITH BOLT ALLOWANCE)



Item	Location	Size (Stainless Steel)	Wt. Lbs.
Valves			(94)
	Engine Bleed Check Valves,	18 req @ 1.5 ea.	27
	Pitch/Yaw Control Valves,	2 req @ 24 lb. ea.	48
	Roll Control Valves	2 req @ 1.5 lb. ea.	19
Fairings			(55)
	Tip Pods	2 req @ 15 lb. ea.	30
	Tail Cone Fairing	1 req @ 25 lb. ea.	25
Total Estimated Weight - Basic Config. Hover Control System			528

Model	Manufacturer	Report Number
T-38A	Northrop-Norair	NOR-66-324
F-100C	North American	Not Available
F-104C	Lockheed	Not Available
F-105D	Republic	EW-97-211
F-106A	Convair	ZW-8-532
F-107A	North American	Not Available
F8U-3	Chance Vought	E8R-11360
F11F-1	Grumman	375-IR
F-5A	Northrop-Norair	NOR-66-56
RF4C	McDonnell	A-560
A4E	Douglas	31388
P-1127	Hawker Aircraft	Issue No. 1

Model	Actual / Estimated	$\bar{X} = 1.000$ $X-\bar{X}$
T-38A	.950	.050
F-100C	1.070	.070
F-104A	1.007	.007
F-105D	1.029	.029
F-106A	.990	.010
F-107A	1.024	.024
F8U-3	.965	.035
F11F-1	1.044	.044
F-5A	.974	.026
RF4C	1.026	.026
A4E	.979	.021
P-1127	.946	.054

$\Sigma \frac{\text{Actual}}{\text{Estimated}} = 12.004$

$\bar{X} = 1.000$

$\Sigma (X-\bar{X})^2 = .016836$

$S = \left[\Sigma(X-\bar{X})^2/(\eta-1) \right]^{.5}$

$S = .03912$

$X-S = 0.96088$
$X+S = 1.03912$

Ref. SAWE Paper No. 529

Confidence Level 90%
$\bar{X} = 1.000$
$S = .03912$
Let $(1-d) = 0.90$
$\therefore d = 0.10$
Since $d = 0.10$, $t = t_{1.0-.05} = t_{0.950}$
$\eta = 12.0$ so $df = 11.0$

Use Table
$t = 0.950$ for $df = 11.0$ is 1.796

Upper Interval $= X_u = \bar{X} + \frac{ts}{N}$

$= 1.00 + \frac{1.796 \times .03912}{12}$

$= 1.00 + .020282$

$\therefore$ Use .020 for contingency

The Accuracy Analysis Basic Data presented in this report utilize the following basic formulae for the individual design group weight estimates.

Wing Group

$$W_w = .008305 \left[(W_g \eta)^{.620} (ARst)^{.590} (S)^{.640} \left(\frac{1+\lambda}{2TR}\right)^{.36} \left(1 + \frac{V}{500}\right)^{.500} \right]^{.914}$$

where,

W_g = Design Gross Weight - lbs.

η = Ultimate Flight Load Factor

ARst = Structural Aspect Ratio - $AR/\cos_4{}^2$

S = Gross Wing Area - sq. ft.

λ = Planform Taper Ratio - Ct/Cr

TR = Thickness Ratio - Per Cent Root Chord

V = Sea Level Limit Speed - Knots

Tail Group

$$W_t = 1.072 \left[W_w \right]^{.842}$$

where,

W_n = Basic Wing Weight - lbs.

Body Group

$$W_b = 4.57 \left[(W_g \eta)^{.597} (B \cdot D)^{.298} (L)^{1.097} \right]^{.516}$$

where,

W_g = Design Gross Weight - lbs.

η = Ultimate Flight Load Factor

B = Maximum Width - ft.

D = Maximum Depth - ft.

L = Fuselage Length - ft.

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Alighting Gear

$$W_g = .05195 \left[(W_g \eta) \right]^{.801}$$

where,

W_g = Design Gross Weight - lbs.

η = Ultimate Flight Load Factor

Surface Controls, Hydraulics & Electrical

$$W_{she} = 2.12 \left[(W_g \eta)^{.623} (L+B)^{.224} (V_{sl})^{.167} \right]^{.654}$$

where,

W_g = Design Gross Weight - lbs.

η = Ultimate Flight Load Factor

L = Fuselage Length - ft.

B = Wing Span - ft.

V_{sl} = Sea Level Limit Speed - Knots

Fuel System

$$W_{fs} = .695 \left[W_f \right]^{.760}$$

where,

W_f = Internal Fuel - lbs.

Accuracy Analysis Basic Data

	T-38A	F-100C	F-104A	F-105D	F-106A	F-107A	F8U-3	F11F-1	F-5A	F-4C	A4E	P-1127
Wing Group												
Actual Weight	788	3,525	1,185	3,529	3,288	3,787	4,128	1,674	1,059	4,670	1,618	1,093
Basic Weight (Calc)	908	2,903	1,169	4,108	3,063	3,906	3,553	1,701	943	3,695	1,023	865
Basic Data												
S	170	385	196	385	697	395	462	255	170	538	260	186
AR	3.75	3.86	2.44	3.18	2.10	3.39	3.46	3.92	3.75	2.74	2.91	2.81
V_{knots}	700	650	760	730	760	700	770	640	700	750	577	620
W_g	9,600	23,996	15,200	34,000	31,377	29,524	30,578	17,500	11,500	37,500	12,504	13,000
$N_{(ult)}$	11.0	11.0	11.0	13.0	10.5	13.0	9.6	9.75	9.8	9.75	10.5	12.0
λ	.200	.262	.377	.467	--	.300	.254	.500	.200	.183	.226	.400
t/Cr	.048	.070	.034	.055	.040	.050	.051	.060	.048	.064	.080	.100
$\Lambda_{c/4}$	25.25°	45°	18.3°	45°	Delta	45°	42°	35°	25.25°	45°	33.2°	35°
Special Features												
Wing Fold							292	33		262		
L.E. Flaps		125		45		90	140		90	395		
Spoilers				37		81	71			60		
Speed Brakes										72		
Int. Fuel Tanks										52	105	
Droops							187					
Catapult											10	
Outrigger												85
Estimated Weight	908	3,028	1,169	4,190	3,063	4,077	4,243	1,734	1,033	4,536	1,138	950
Tail Group												
Actual Weight	300	952	447	991	703	1,130	1,045	670	300	926	615	305
Estimated Weight	294	1,009	415	1,020	703	1,062	1,018	546	350	1,115	507	362

Accuracy Analysis Basic Data

	T-38A	F-100C	F-104A	F-105D	F-106A	F-107A	F8U-3	F11F-1	F-5A	F-4C	A4E	P-1127
Body Group												
Actual Weight	2,090	3,560	2,670	5,918	4,508	4,792	3,850	3,269	2,172	4,946	1,434	1,674
Basic Weight (Calc)	2,272	3,156	2,845	4,522	4,323	4,180	3,964	2,683	2,342	4,380	2,290	2,607
Basic Data												
W_g	9,600	23,996	15,200	34,000	31,377	29,524	30,578	17,500	11,500	37,500	12,504	13,000
N(ult)	11.0	11.0	11.0	13.0	10.5	13.0	9.6	9.75	9.8	9.75	10.5	12.0
B - width - ft.	5.2	5.6	5.5	4.38	8.1	5.65	6.03	5.58	5.2	6.3	5.3	6.0
D - depth - ft.	5.0	5.6	4.7	7.41	6.5	7.36	7.68	5.25	5.0	7.8	5.0	5.6
L - length - ft.	44.2	45.6	51.3	64.4	61.2	56.6	57.9	44.2	45.0	60.3	39.6	42.5
Special Features												
Armor Plate		48										
Bomb Bay				842	508							
Aux. Gear							197	179				
Photo										174		
Ram Air										49		
Gear - Catapult, Arrest										194		
Estimated Weight	2,272	3,204	2,845	5,364	4,831	4,180	4,161	2,862	2,342	4,797	2,290	2,607
Alighting Gear Group												
Actual Weight	517	1,365	844	2,142	1,235	1,381	948	907	613	1,960	819	732
Basic Weight (Calc)	548	1,144	793	1,728	1,366	1,543	1,295	806	578	1,484	654	748

Accuracy Analysis Basic Data

	T-38A	F-100C	F-104A	F-105D	F-106A	F-107A	F8U-3	F11F-1	F-5A	F-4C	A4E	P-1127
Surface Controls, Hydraulics & Electrical												
Actual Weight	867	1,416	1,327	2,671	1,472	2,051	2,013	1,385	929	2,198	916	1,143
Basic Weight (Calc)	899	1,332	1,102	1,706	1,520	1,588	1,449	1,096	925	1,582	957	1,031
Basic Data												
W_g	9,600	24,000	15,200	34,000	31,377	29,524	30,578	17,500	11,500	37,500	12,504	13,000
N(ult)	11.0	11.0	11.0	13.0	10.5	13.0	9.6	9.75	9.8	9.75	10.5	12.0
L - length - ft.	44.2	45.6	51.3	64.4	61.2	56.6	57.9	44.2	45.0	60.3	39.6	42.5
B - span - ft.	25.3	38.6	21.9	35.0	38.3	36.6	40.0	31.6	25.3	38.4	27.5	22.9
V_{sl} - knots	700	650	760	730	760	700	770	640	700	750	577	620
Special Features												
Leading Edge Flaps			61	199			287		40	150		
Stab. Aug.				146		150				316	31	
Spoilers				180		253	71	35		167		
Incidence							100					
Misc.				107		77	49	109				
Estimated Weight	899	1,332	1,163	2,338	1,520	2,068	1,956	1,240	965	2,215	988	1,031
Fuel System												
Actual Weight	333	786	451	777	792	983	848	463	333	1,084	442	455
Basic Weight (Calc)	366	630	449	617	755	633	970	561	366	891	481	315
Basic Data												
Int. Fuel Weight	3,790	7,768	4,960	7,540	9,841	7,800	13,676	6,663	3,790	12,278	5,440	4,620

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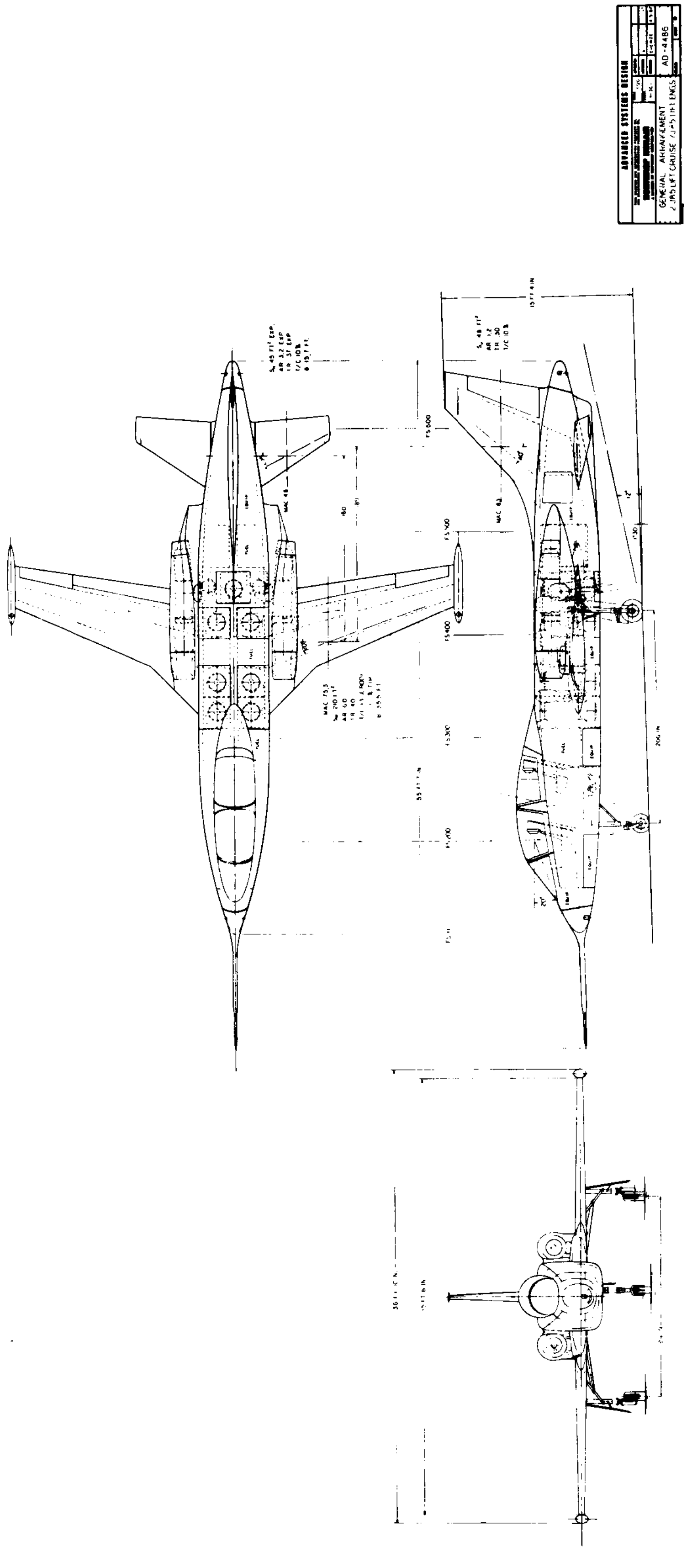
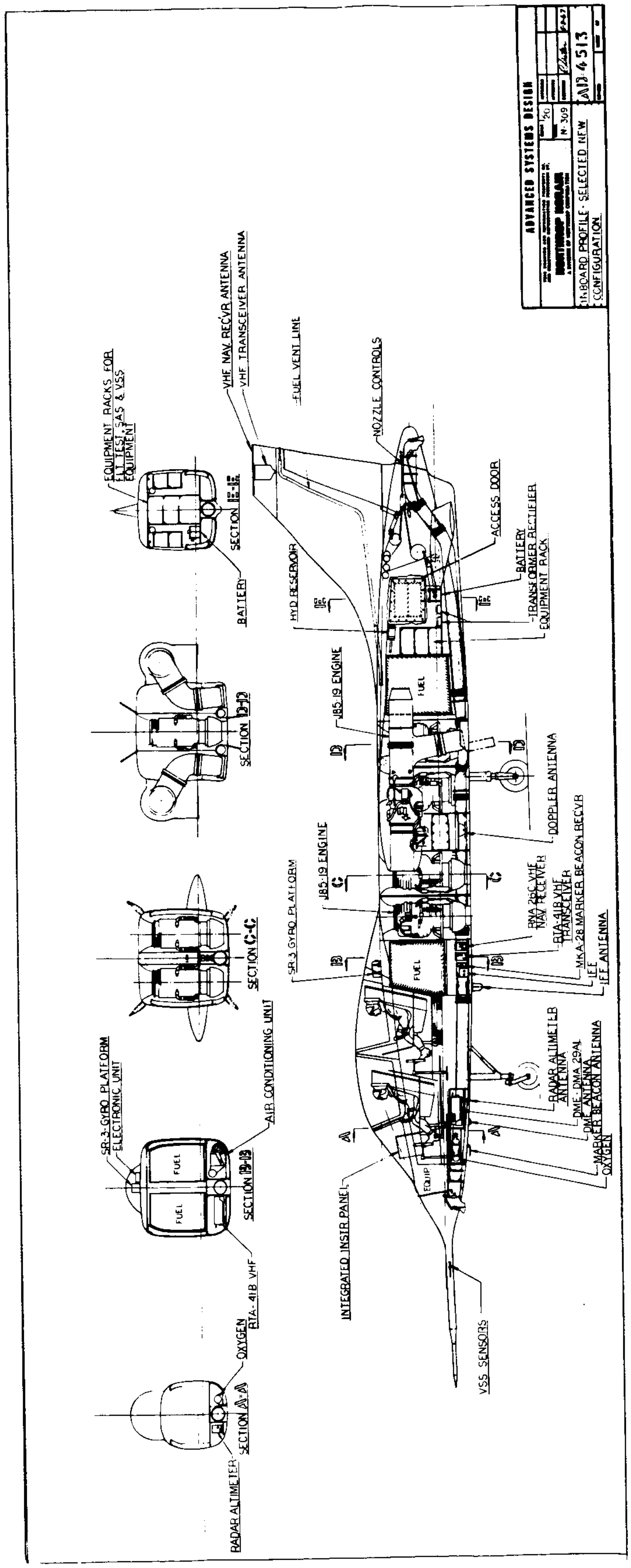
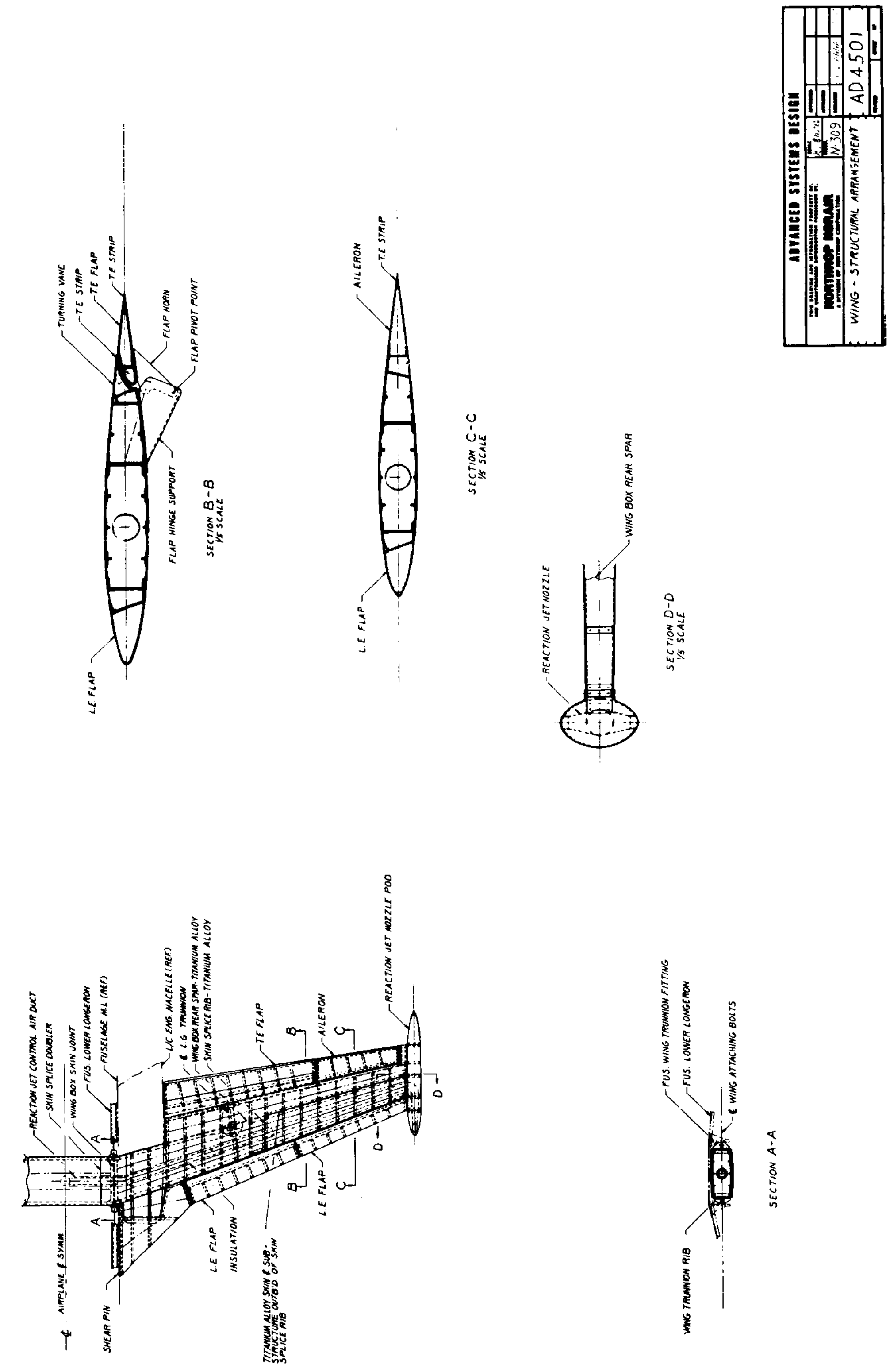
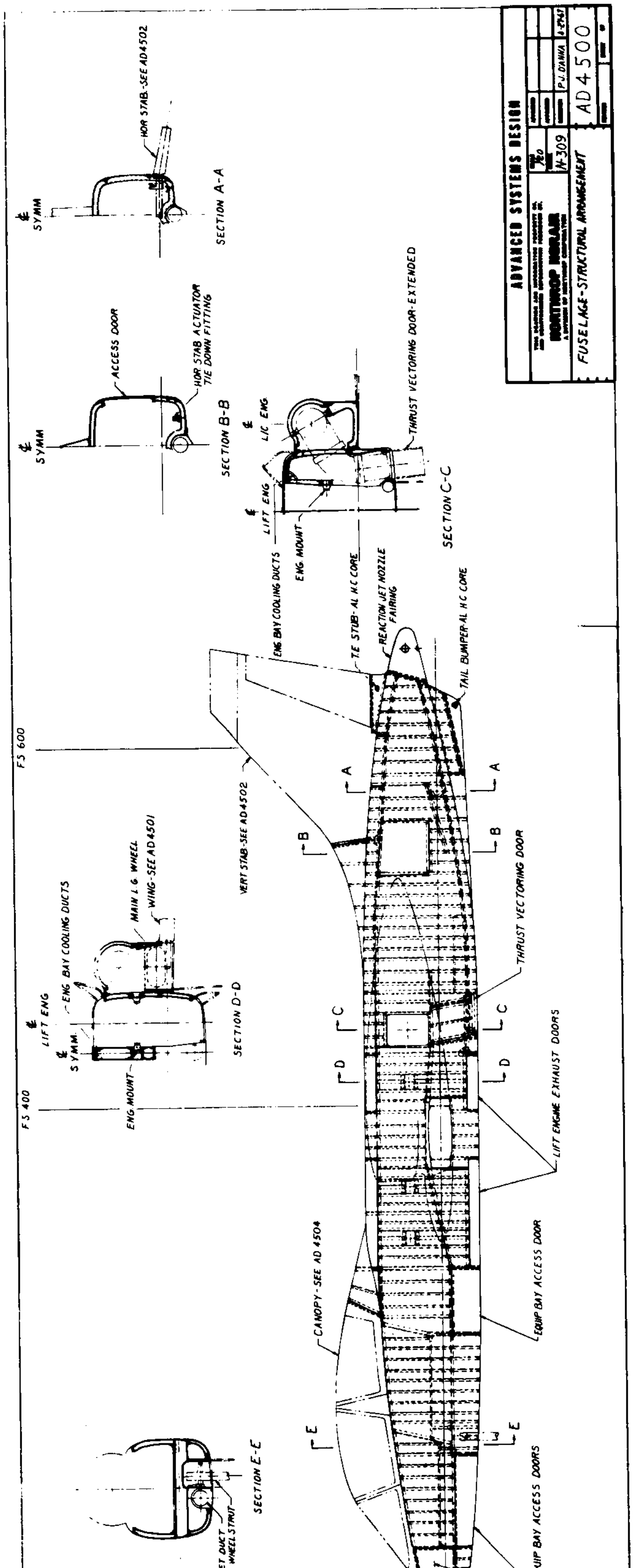
DATE	Contingency Analysis AD-4486A	MODEL N-309

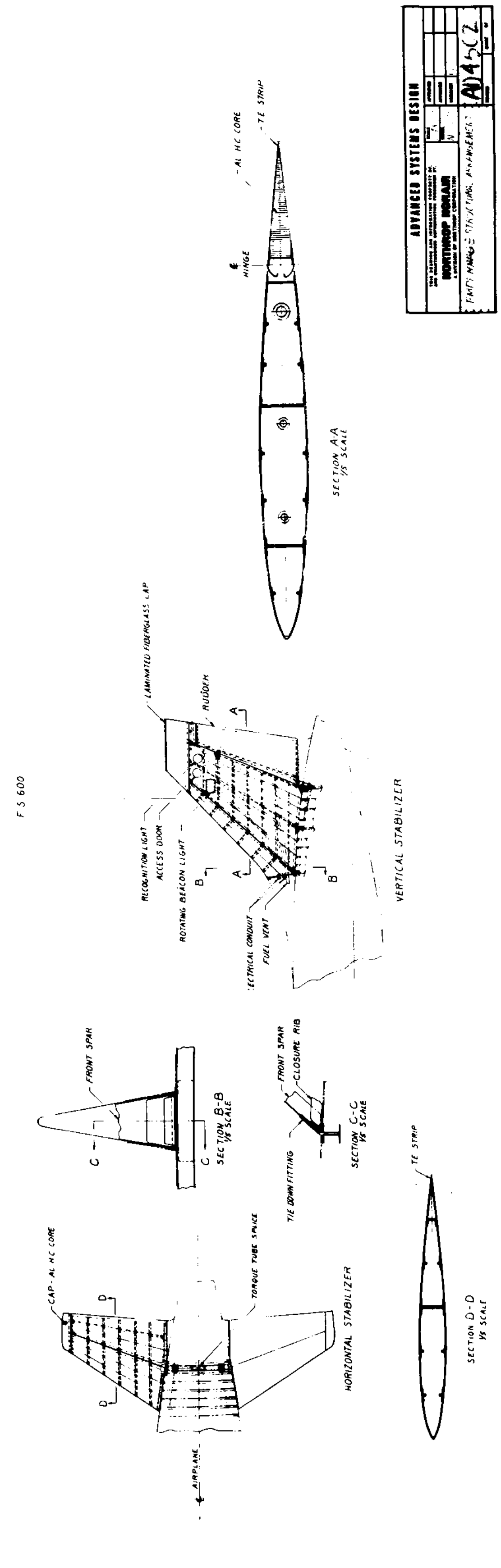
AN-91030 Group Weight Statement	Accuracy Analysis 2%	Other Weight 5%	Accountable Weight	AD-4486A Estimated Weight
Wing Group	833	97		930
Tail Group	309	--		309
Body Group	2,280	257		2,537
Alighting Gear Group	531	261		792
Surface Controls Group	325	183		508
Engine Section		680		680
Propulsion Group	375	491	3,874	4,740
Instrument Group		171		171
Hydraulic Group	190			190
Electrical Group	335			335
Electronics Group		61	159	220
Furn. & Equip. Group		410		410
Air Cond. Group		68		68
Hovering Controls		528		528
Weight Empty	5,178	3,207	4,033	12,418
Useful Load				
Crew			400	400
Engine Oil			63	63
Unusable Fuel		45		45
Operating Weight Empty	5,178	3,252	4,496	12,926
Contingency	104	163	--	267

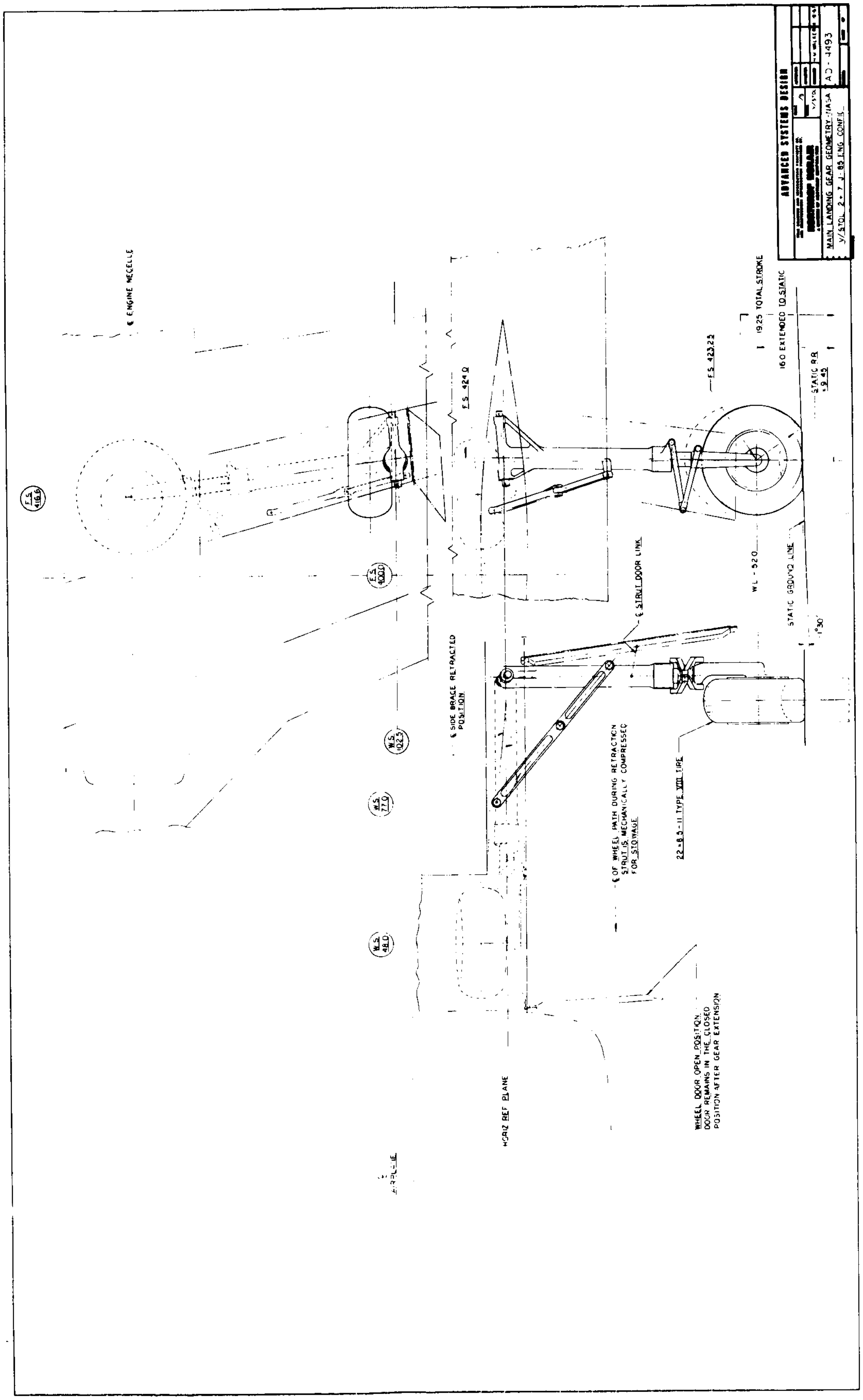

ADVANCED SYSTEMS DESIGN
GENERAL ARRANGEMENT
AD-4486

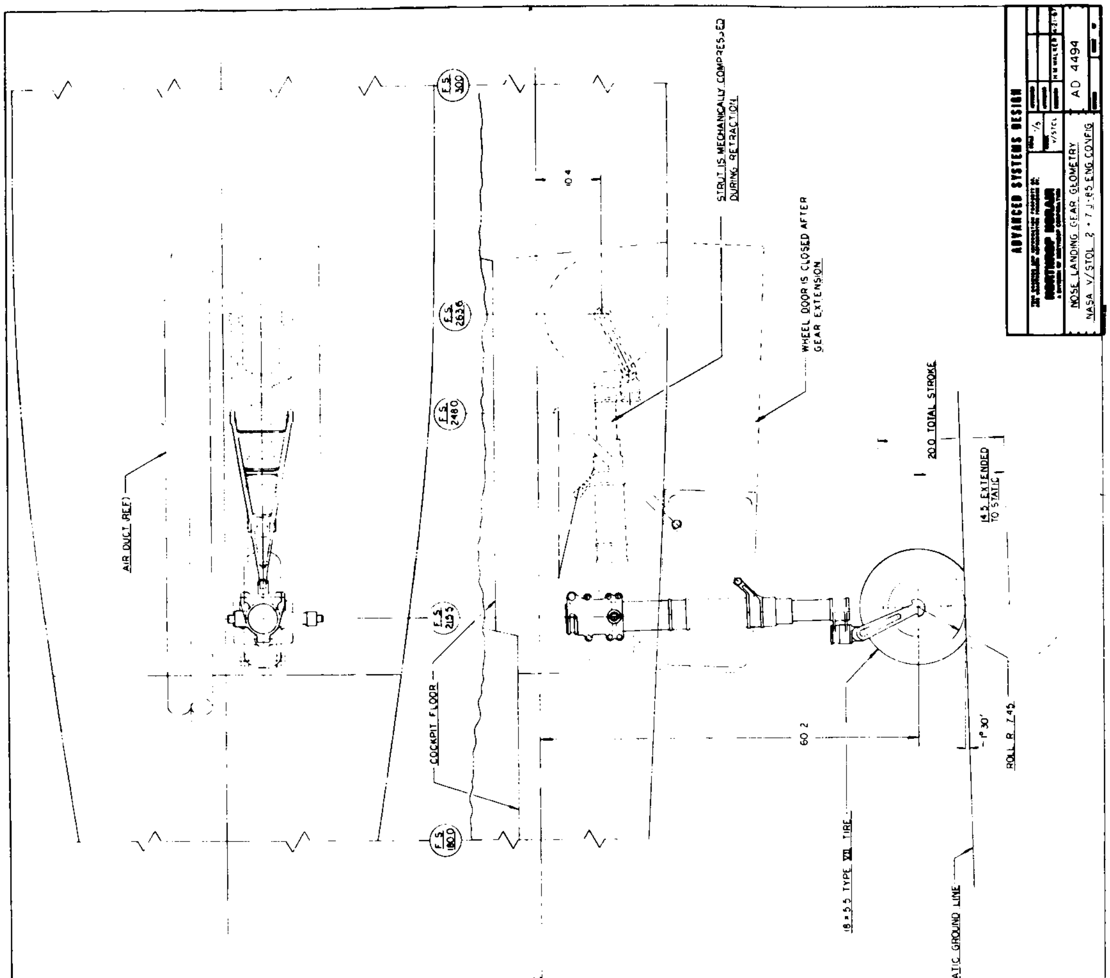

ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
IN-BOARD PROFILE - SELECTED NEW CONFIGURATION
A13 4513
EQUIPMENT RACKS FOR FLT TEST, SAS & VSS EQUIPMENT
VHF NAV RECVR ANTENNA
VHF TRANSCEIVER ANTENNA
FUEL VENT LINE
NOZZLE CONTROLS
ACCESS DOOR
BATTERY
TRANSFORMER RECTIFIER
EQUIPMENT RACK
SECTION E-E
BATTERY
HYD RESERVOIR
J85-19 ENGINE
SECTION D-D
DOPPLER ANTENNA
J85-19 ENGINE
RNA 26C VHF NAV RECEIVER
RTA-41B VHF TRANSCEIVER
MKA-28 MARKER BEACON RECVR
IFF
IFF ANTENNA
SR-3 GYRO PLATFORM
SECTION C-C
FUEL
AIR CONDITIONING UNIT
SR-3 GYRO PLATFORM ELECTRONIC UNIT
RADAR ALTIMETER ANTENNA
DME-DMA 29AL
DME ANTENNA
MARKER BEACON ANTENNA
OXYGEN
SECTION B-B
FUEL
EQUIP
INTEGRATED INSTR PANEL
RTA-41B VHF
OXYGEN
VSS SENSORS
SECTION A-A
RADAR ALTIMETER

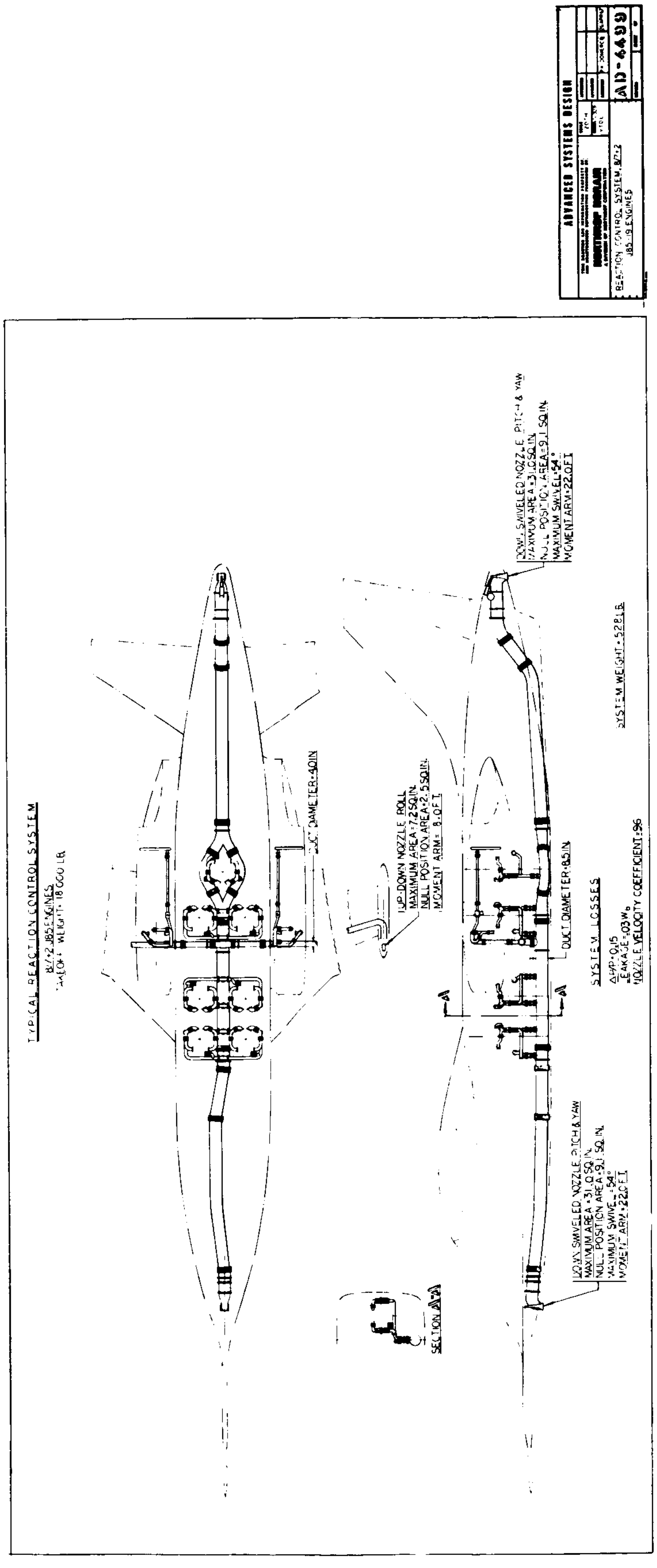

TURNING VANE
TE STRIP
TE FLAP
TE STRIP
FLAP HORN
FLAP PIVOT POINT
FLAP HINGE SUPPORT
LE FLAP
SECTION B-B
1/5 SCALE
AILERON
TE STRIP
LE FLAP
SECTION C-C
1/5 SCALE
REACTION JET NOZZLE
WING BOX REAR SPAR
SECTION D-D
1/5 SCALE
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
N-309
WING - STRUCTURAL ARRANGEMENT
AD 4501
REACTION JET CONTROL AIR DUCT
SKIN SPLICE DOUBLER
WING BOX SKIN JOINT
FUS. LOWER LONGERON
FUSELAGE M.L. (REF)
L/C ENG NACELLE (REF)
℄ L.G. TRUNNION
WING BOX REAR SPAR-TITANIUM ALLOY
SKIN SPLICE RIB-TITANIUM ALLOY
TE FLAP
AILERON
REACTION JET NOZZLE POD
AIRPLANE ℄ SYMM.
SHEAR PIN
LE FLAP
INSULATION
TITANIUM ALLOY SKIN & SUB-STRUCTURE OUTB'D OF SKIN SPLICE RIB
FUS WING TRUNNION FITTING
FUS. LOWER LONGERON
℄ WING ATTACHING BOLTS
WING TRUNNION RIB
SECTION A-A


ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
N-309
FUSELAGE-STRUCTURAL ARRANGEMENT
AD4500
SECTION A-A
HOR STAB-SEE AD4502
SYMM
SECTION B-B
ACCESS DOOR
HOR STAB ACTUATOR TIE DOWN FITTING
SECTION C-C
LIFT ENG
L/C ENG
ENG MOUNT
THRUST VECTORING DOOR-EXTENDED
ENG BAY COOLING DUCTS
T.E STUB-AL H C CORE
REACTION JET NOZZLE FAIRING
TAIL BUMPER-AL H C CORE
FS 600
VERT STAB-SEE AD4502
THRUST VECTORING DOOR
SECTION D-D
ENG BAY COOLING DUCTS
MAIN L G WHEEL
WING-SEE AD4501
FS 400
LIFT ENGINE EXHAUST DOORS
CANOPY-SEE AD 4504
EQUIP BAY ACCESS DOOR
SECTION E-E
REACTION JET DUCT
NOSE L G WHEEL STRUT
EQUIP BAY ACCESS DOORS
FS 111


ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
AL HC CORE
TE STRIP
HINGE
SECTION A-A 1/5 SCALE
LAMINATED FIBERGLASS LAP
RUDDER
F S 600
RECOGNITION LIGHT
ACCESS DOOR
ROTATING BEACON LIGHT
ELECTRICAL CONDUIT
FUEL VENT
VERTICAL STABILIZER
FRONT SPAR
SECTION B-B 1/5 SCALE
FRONT SPAR
CLOSURE RIB
TIE DOWN FITTING
SECTION C-C 1/5 SCALE
TE STRIP
CAP-AL HC CORE
TORQUE TUBE SPLICE
HORIZONTAL STABILIZER
SECTION D-D 1/5 SCALE
AIRPLANE


ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
MAIN LANDING GEAR GEOMETRY-NASA
V/STOL 2+2 J-85 ENG CONFIG
AD-4493
℄ ENGINE NACELLE
F.S. 416.6
F.S. 424.0
F.S. 423.22
19.25 TOTAL STROKE
16.0 EXTENDED TO STATIC
STATIC R.R. 19.45
WL - 52.0
STATIC GROUND LINE
1°30'
℄ STRUT DOOR LINK
F.S. 400.0
℄ SIDE BRACE RETRACTED POSITION
W.S. 102.5
W.S. 77.0
℄ OF WHEEL PATH DURING RETRACTION STRUT IS MECHANICALLY COMPRESSED FOR STOWAGE
22 x 6.5-11 TYPE VIII TIRE
W.S. 48.0
HORIZ REF PLANE
WHEEL DOOR OPEN POSITION DOOR REMAINS IN THE CLOSED POSITION AFTER GEAR EXTENSION
℄ AIRPLANE


STRUT IS MECHANICALLY COMPRESSED DURING RETRACTION
WHEEL DOOR IS CLOSED AFTER GEAR EXTENSION
20.0 TOTAL STROKE
14.5 EXTENDED TO STATIC
AIR DUCT REF
COCKPIT FLOOR
F.S. 300
F.S. 263.6
F.S. 248.0
F.S. 215.5
F.S. 180.0
60.2
ROLL R 7.45
18 x 5.5 TYPE VII TIRE
STATIC GROUND LINE
HORIZ REF PLANE
A-4 E NOSE GEAR STRUT & WHEEL ASSEMBLY MODIFIED
AIRPLANE
6.0
ADVANCED SYSTEMS DESIGN
NOSE LANDING GEAR GEOMETRY
AD 4494


ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
AD-6499
TYPICAL REACTION CONTROL SYSTEM
UP-DOWN NOZZLE ROLL
MAXIMUM AREA 7.25 SQ IN
NULL POSITION AREA 2.5 SQ IN
MOMENT ARM 8.0 FT
DUCT DIAMETER 8.5 IN
SYSTEM LOSSES
SYSTEM WEIGHT 528 LB
DOWN SWIVELED NOZZLE PITCH & YAW
MAXIMUM AREA 31.0 SQ IN
NULL POSITION AREA 9.1 SQ IN
MAXIMUM SWIVEL 54°
MOMENT ARM 22.0 FT
SECTION A-A

Section II

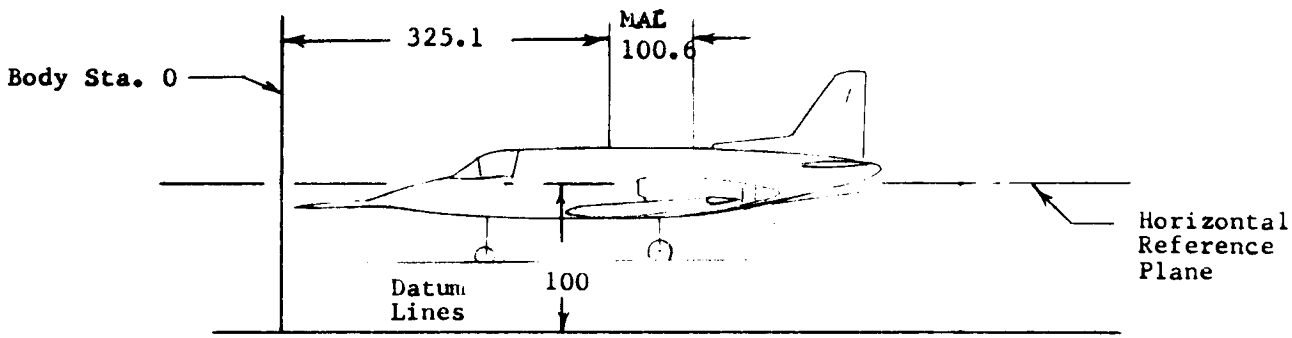
Modified Airplane
T-39A

Drawing No. AD-4448A

(See Page 1-13 of this report
or Page 5-II-113 of this section)

Condition	Weight, Pounds	Center of Gravity: Horizontal Fus. Sta.	Center of Gravity: Horizontal % MAC	Center of Gravity: Vertical Water Line	Moment of Inertia, Slug-Ft^2: Roll Ixx	Moment of Inertia, Slug-Ft^2: Pitch Iyy	Moment of Inertia, Slug-Ft^2: Yaw Izz
Composite Mode:							
Gross Wt. Less Fuel	15,966	338.8	13.6	93.8	15,021	40,417	49,025
Usable Fuel	4,234	335.4	10.2	105.0			
VTOL Gross Weight	20,200	338.1	12.9	96.2	15,112	49,616	58,133
Direct Lift Mode:							
Gross Wt. Less Fuel	16,378	338.5	13.3	94.1	14,948	39,300	48,017
Usable Fuel	3,822	336.0	10.8	105.0			
VTOL Gross Weight	20,200	338.0	12.8	96.2	15,028	48,428	57,065

NOTE: Above conditions for landing gear down and a crew of two. Fuselage stations shown are Norair designations.



	Direct Lift	Lift/Cruise	AD-4448A Cat. II Composite T-39A Modified	Original T-39A NA 63-1347
Structure			(5,750)	(4,381)
Wing			1,795	1,661
Tail			311	311
Body (incl. air induction & exhaust doors)			2,555	1,780
Alighting Gear			1,089	629
Propulsion & Nacelle	(3,649)	(2,438)	(6,087)	(1,644)
Engine Installation	3,096	784		980
Swivel Nozzle/Diverter Valve	264	150		--
Engine Section, Nacelles & Pylons	146	745		343
Air Induction	40	36		29
Exhaust & Cooling	16	138		44
Lubrication System	7	2		1
Engine Controls	50	28		30
Starting System	30	15		22
Constant Speed Drives	--	140		--
Fuel System	--	400		195
Power Systems			(1,658)	(1,395)
Surface Controls			508	326
Hydraulics			190	145
Electrical			335	924
Hover Controls (ducting & valves)			625	--
Equipment Groups			(869)	(1,843)
Instruments			171	166
Electronics			220	464
Furnishings & Equipment			410	877
Air Conditioning & Anti-Icing			68	333
Auxiliary Gear			--	3
Contingency			(287)	--
Weight Empty			14,651	9,263
Operating Weight Empty Items			(515)	(534)
Crew (2)			400	340
Unusable & Trapped Fuel			45	170
Oil			70	24
Operating Weight Empty			15,166	9,797
Usable Fuel			4,234	6,863
Payload			800	1,100
Maximum VTOL/Gross Weight			20,200	17,760
Limit Load Factor (4.0 at 16,527 lbs.)			3.27	3.72

NAME
DATE

AD-4448A
Composite Mode
T-39A
Modified Airplane

GROUP WEIGHT STATEMENT

ESTIMATED - XXXXXXXXXXXXXXXXXX

(Cross out those not applicable)

CONTRACT NO.

AIRPLANE, GOVERNMENT NO.

AIRPLANE, CONTRACTOR NO.

MANUFACTURED BY

		MAIN	AUXILIARY-Lift
ENGINE	MANUFACTURED BY	General Electric	General Electric
	MODEL	J85-19	J85-19
	NO.	2	8
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

AN-9103-D
NAME
DATE

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 98
MODEL T-39 Mod.
REPORT

No.	Item					Group
1	WING GROUP					1,795
2	CENTER SECTION - BASIC STRUCTURE				400	
3	INTERMEDIATE PANEL - BASIC STRUCTURE					
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS LBS.)				1,096	
5						
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM LBS.)				37	
7	AILERONS (INCL. BALANCE WEIGHT 9 LBS.)				56	
8	FLAPS - TRAILING EDGE				94	
9	- LEADING EDGE					
10	SLATS				112	
11	SPOILERS					
12	SPEED BRAKES					
13						
14						
15	TAIL GROUP					311
16	STABILIZER - BASIC STRUCTURE H				136	
17	FINS - BASIC STRUCTURE (INCL. DORSAL LBS.) V				55	
18	SECONDARY STRUCTURE (STAB. & FINS) H = 8 V = 3				11	
19	ELEVATOR (INCL. BALANCE WEIGHT 21 LBS.) H				78	
20	RUDDERS (INCL. BALANCE WEIGHT 6 LBS.) V				31	
21						
22						
23	BODY GROUP					2,555
24	FUSELAGE OR HULL - BASIC STRUCTURE				1,940	
25	BOOMS - BASIC STRUCTURE					
26	SECONDARY STRUCTURE - FUSELAGE OR HULL				152	
27	- BOOMS					
28	- SPEEDBRAKES					
29	- DOORS, PANELS & MISC.				463	
30						
31	ALIGHTING GEAR GROUP - LAND (TYPE: Tricycle)					1,089
32-33	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS		
34	Main - Nacelles	260	372	63	695	
35	Nose - Fuselage	26	134	34	194	
36	Main - Heat Shield		200		200	
37						
38						
39						
40	ALIGHTING GEAR GROUP - WATER					
41	LOCATION	FLOATS	STRUTS	CONTROLS		
42						
43						
44						
45						
46	SURFACE CONTROLS GROUP					508
47	COCKPIT CONTROLS				68	
48	Stability Augmentation System				118	
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS LBS.)				322	
50						
51	ENGINE SECTION OR NACELLE GROUP					891
52	INBOARD					
53	CENTER - Lift Engines				216	
54	OUTBOARD - Lift/Cruise Engines				675	
55	DOORS, PANELS & MISC.					
56						
57	TOTAL (TO BE BROUGHT FORWARD)					7,149

AN-9103-D
NAME
DATE

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 99
MODEL T-39 Mod.
REPORT

#	Item		Auxiliary-Lift		Main-Cruise	
1	PROPULSION GROUP					5,196
2			AUXILIARY-Lift		MAIN-Cruise	
3	ENGINE INSTALLATION		3,096		784	
4	AFTERBURNERS (IF FURN. SEPARATELY)					
5	ACCESSORY GEAR BOXES & DRIVES - CSD				140	
6	SUPERCHARGERS (FOR TURBO TYPES)					
7	AIR INDUCTION SYSTEM		40		36	
8	EXHAUST SYSTEM & Cooling		16		138	
9	~~XXXXXXXXXX~~ Swivel Nozzle/Diverters		264		150	
10	LUBRICATING SYSTEM		7		2	
11	TANKS					
12	COOLING INSTALLATION					
13	DUCTS, PLUMBING, ETC.	7		2		
14	FUEL SYSTEM				400	
15	TANKS - PROTECTED					
16	- UNPROTECTED			97		
17	PLUMBING, ETC.			303		
18	WATER INJECTION SYSTEM					
19	ENGINE CONTROLS		50		28	
20	STARTING SYSTEM		30		15	
21	PROPELLER INSTALLATION					
22						
23						
24	AUXILIARY POWER PLANT GROUP					
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP					171
26	HYDRAULIC & PNEUMATIC GROUP					190
27	Hover Control System (ducting & valves)					625
28						
29	ELECTRICAL GROUP					335
30						
31						
32	ELECTRONICS GROUP					220
33	EQUIPMENT				174	
34	INSTALLATION				46	
35						
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)					
37	FURNISHINGS & EQUIPMENT GROUP					410
38	ACCOMMODATIONS FOR PERSONNEL				305	
39	MISCELLANEOUS EQUIPMENT				44	
40	FURNISHINGS				26	
41	EMERGENCY EQUIPMENT				35	
42						
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP					68
44	AIR CONDITIONING & Equipment Cooling				61	
45	ANTI-ICING - Cabin De-Fog				7	
46						
47	PHOTOGRAPHIC GROUP					
48	AUXILIARY GEAR GROUP					
49	HANDLING GEAR					
50	ARRESTING GEAR					
51	CATAPULTING GEAR					
52	ATO GEAR					
53						
54						
55	~~XXXXXXXXXXXXXXXXXXXX~~ Contingency					287
56	TOTAL FROM PG. 2					7,149
57	WEIGHT EMPTY					14,651

GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

				Composite Mode		Direct Lift Mode	
1 LOAD CONDITION							
2							
3 CREW (NO. 2)				400		400	
4 PASSENGERS (NO.)							
5 FUEL	Type	Gals.					
6 UNUSABLE	JP-4	6.9		45		45	
7 INTERNAL- Flight	JP-4	651.4		4,234		--	
8 - Flight	JP-4	588.0		--		3,822	
9							
10 EXTERNAL							
11							
12 BOMB BAY							
13							
14 OIL				70		84	
15 TRAPPED							
16 ENGINE							
17							
18 FUEL TANKS (LOCATION)							
19 WATER INJECTION FLUID (GALS)							
20							
21 BAGGAGE							
22 CARGO							
23							
24 ARMAMENT							
25 GUNS (Location)	Fix. or Flex.	Qty.	Cal.				
26							
27							
28							
29							
30							
31							
32 AMMUNITION							
33							
34							
35							
36							
37							
38							
39 INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)							
*40 BOMB OR TORPEDO RACKS							
41							
42 Variable Stability System				500		500	
43 NASA Research Equipment				300		--	
44							
45							
46 EQUIPMENT							
47 PYROTECHNICS							
48 PHOTOGRAPHIC							
49							
*50 OXYGEN							
51							
52 MISCELLANEOUS							
53							
54							
55 USEFUL LOAD				5,549		4,851	
56 WEIGHT EMPTY				14,651		15,349	
57 GROSS WEIGHT				20,200		20,200	

*If not specified as weight empty.

GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

1 LENGTH - OVERALL (FT.) 55.5 — HEIGHT - OVERALL - STATIC (FT.) 17.2

No.	Item	Main Floats	Aux. Floats	Booms	Fuse or Hull	Nacelles Inboard	Nacelles Center	Nacelles Outboard
3	LENGTH - MAX. (FT.)				47.00	13.75		
4	DEPTH - MAX. (FT.)				6.42	2.42		
5	WIDTH - MAX. (FT.)				5.75	2.58		
6	WETTED AREA (SQ. FT.)				670.00	188.00		
*7	FLOAT OR HULL DISPL. - MAX (LBS.)							

8 FUSELAGE VOLUME (CU. FT.) — PRESSURIZED — TOTAL

No.	Item	Wing	H. Tail	V. Tail
10	GROSS AREA (SQ. FT.)	342.00	77.00	41.58
11	WEIGHT/GROSS AREA (LBS./SQ. FT.)	5.25	2.88	2.14
12	SPAN (FT.)	44.64	17.55	7.86
13	FOLDED SPAN (FT.)			
14				
15	SWEEPBACK - AT 25% CHORD LINE (DEGREES)	28.55	30.00	30.00
16	- AT % CHORD LINE (DEGREES)			
**17	THEORETICAL ROOT CHORD - LENGTH (INCHES)	139.86	81.22	95.69
18	- MAX. THICKNESS (INCHES)	15.81	8.11	9.44
**19	CHORD AT PLANFORM BREAK - LENGTH (INCHES)			
20	- MAX. THICKNESS (INCHES)			
***21	THEORETICAL TIP CHORD - LENGTH (INCHES)	44.91	24.37	28.32
22	- MAX. THICKNESS (INCHES)	4.20	2.43	2.83

23 DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.) 6.13
24 TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.) 17.32

No.	AREAS (SQ. FT.) (Per APN)			
25	Flaps	L.E.	T.E. 40.26	
26	Lateral Controls	Slats 36.34	Spoilers	Ailerons 16.42
27	Speed Brakes	Wing	Fuse. or Hull	
28	Elevators			16.52
29	Rudder			8.95

No.	ALIGHTING GEAR (LOCATION)	Nose	Main		
31	LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)	65.90	53.40		
32	OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)	17.90	14.00		
33	FLOAT OR SKI STRUT LENGTH (INCHES)				

34 ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)
35 HYDRAULIC SYSTEM CAPACITY (GALS.)

No.	FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
37	Fuel - Internal	Wing				
38		Fuse. or Hull				
39	- External					
40	- Bomb Bay					
41						
42	Oil					
43						

44

No.	STRUCTURAL DATA - CONDITION	Fuel In Wings (Lbs.)	Stress Gross Weight	Ult. L.F.
46	FLIGHT	0	16,527	6.0
47	LANDING			
48				
49	MAX. GROSS WEIGHT WITH ZERO WING FUEL			
50	CATAPULTING			
51	MIN. FLYING WEIGHT			
52	LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)			

53 WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)
54 STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)
55 PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)
56
57 AIRFRAME WEIGHT (AS DEFINED IN ~~XXXXXX~~ AMPR (LBS.) 9,553

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ at ℄ airplane.
***Parallel to ℄ airplane.
****Total usable capacity.

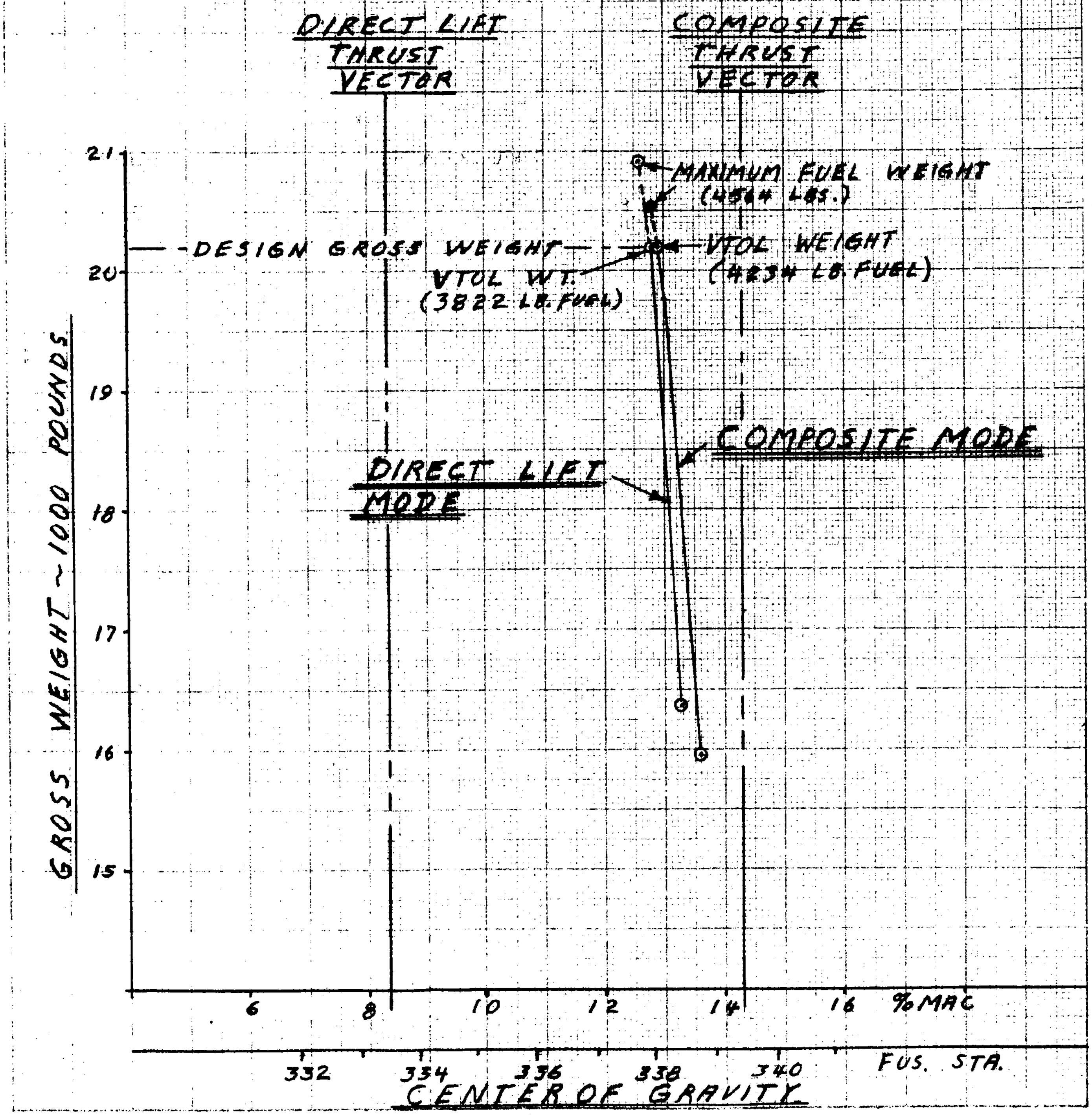
ESTIMATED AMPR WEIGHT

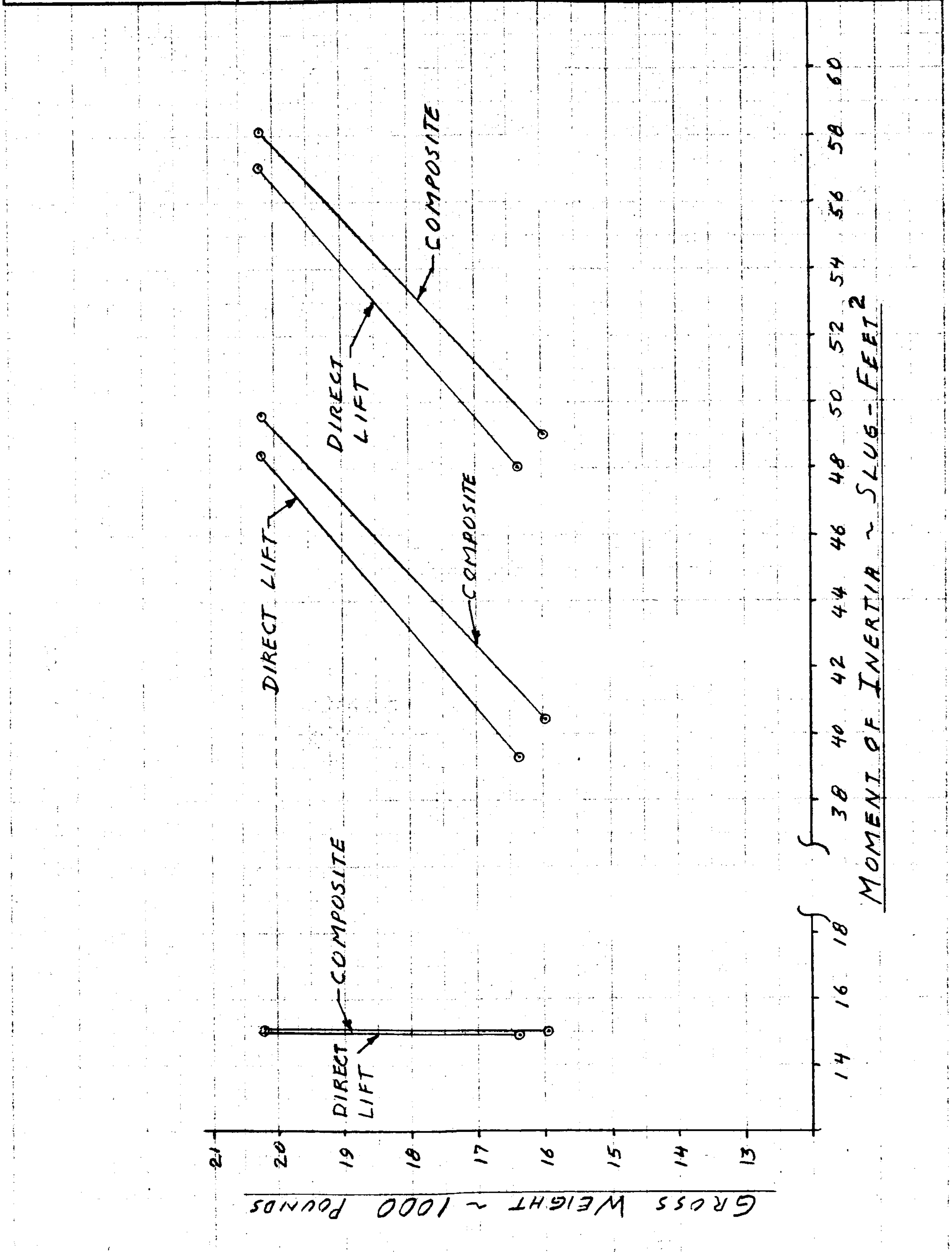
Group	Original T-39A AMPR Weight	Δ AMPR Wt. Wt. Out	Δ AMPR Wt. Wt. In	Modified AMPR Wt.	Modified Empty Weight
Wing	1,661	- 272	406	1,795	1,795
Tail	311	--	--	311	311
Body	1,780	- 956	1,731	2,555	2,555
Alighting Gear	420	- 420	803	803	1,089
Surface Controls	326	- 326	508	508	508
Engine Section & Nacelles	343	- 343	891	891	891
Propulsion	321	- 321	805	805	5,196
Instruments	77	- 77	54	54	171
Hydraulics	145	- 145	190	190	190
Electrical	556	- 556	215	215	335
Electronics	139	- 139	61	61	220
Furnishings & Equipment	877	- 877	410	410	410
Air Conditioning	289	- 289	43	43	68
Auxiliary Gear	3	- 3	--	--	--
Hover Controls	--	--	625	625	625
Empty Weight	7,248	-4,724	6,742	9,266	14,364
Contingency	--	--	287	287	287
Empty + Contingency	7,248	-4,724	7,029	9,553	14,651

ENGINEER	NORTHROP CORPORATION NORAIR DIVISION	PAGE 103
CHECKER		REPORT NO.
DATE	CENTER OF GRAVITY DIAGRAM	MODEL T-39 MOD.

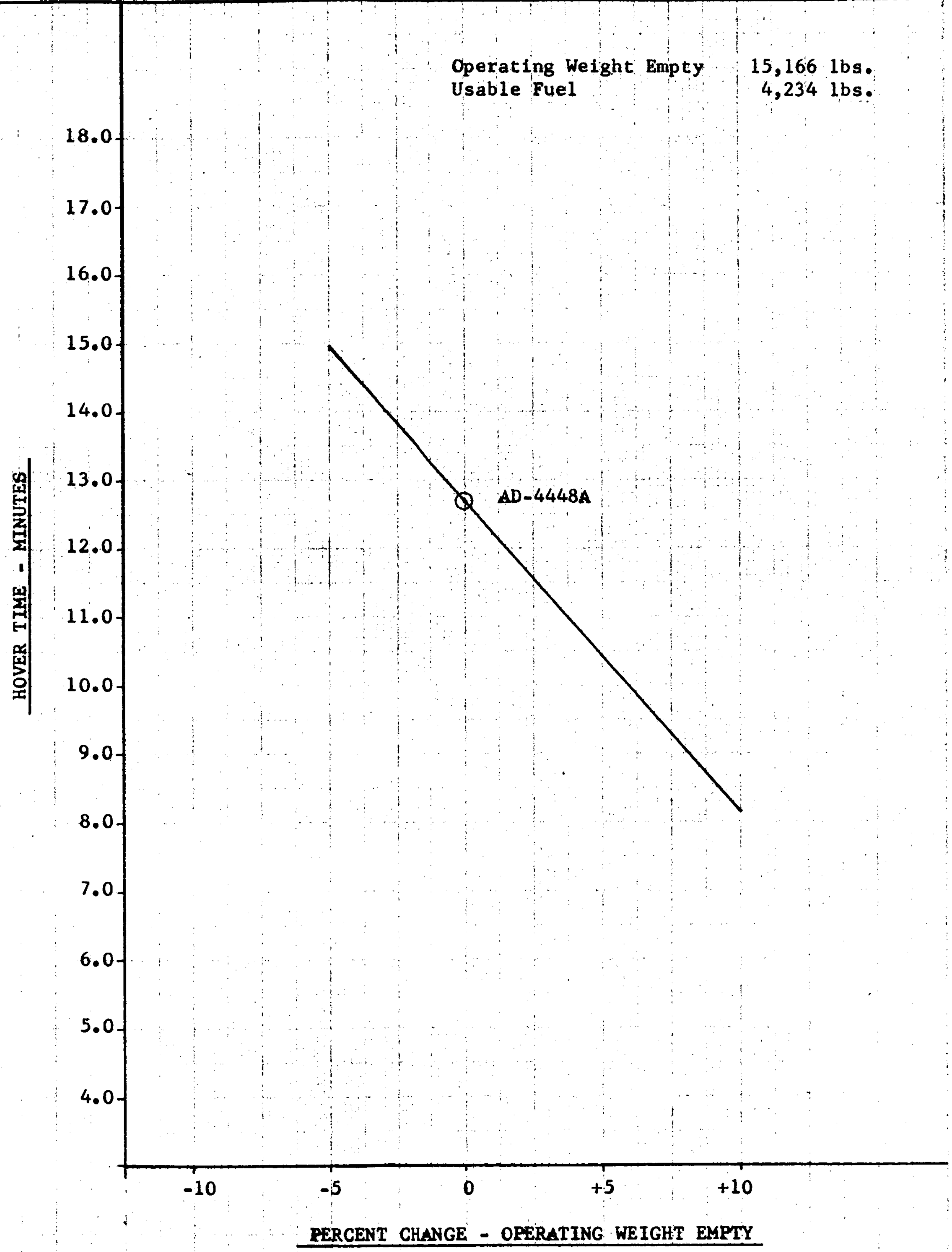
NOTES

1. GEAR DOWN CONDITION SHOWN. EFFECT OF GEAR RETRACTION IS APPROXIMATELY +1.5 % MAC AT 16,000 LBS. AND +1.2 % MAC AT 20,200 LBS.
2. TWO MAN CREW CONDITION SHOWN. EFFECT OF ONE MAN OUT IS APPROX. +1.9 % MAC AT 16,000 LBS. AND +1.4 % MAC AT 20,200 LBS.
3. MAXIMUM AFT C.G. IN FLIGHT IS 31 % MAC. THE FORWARD LIMIT IS TO BE DETERMINED BY STRUCTURAL LIMITS.



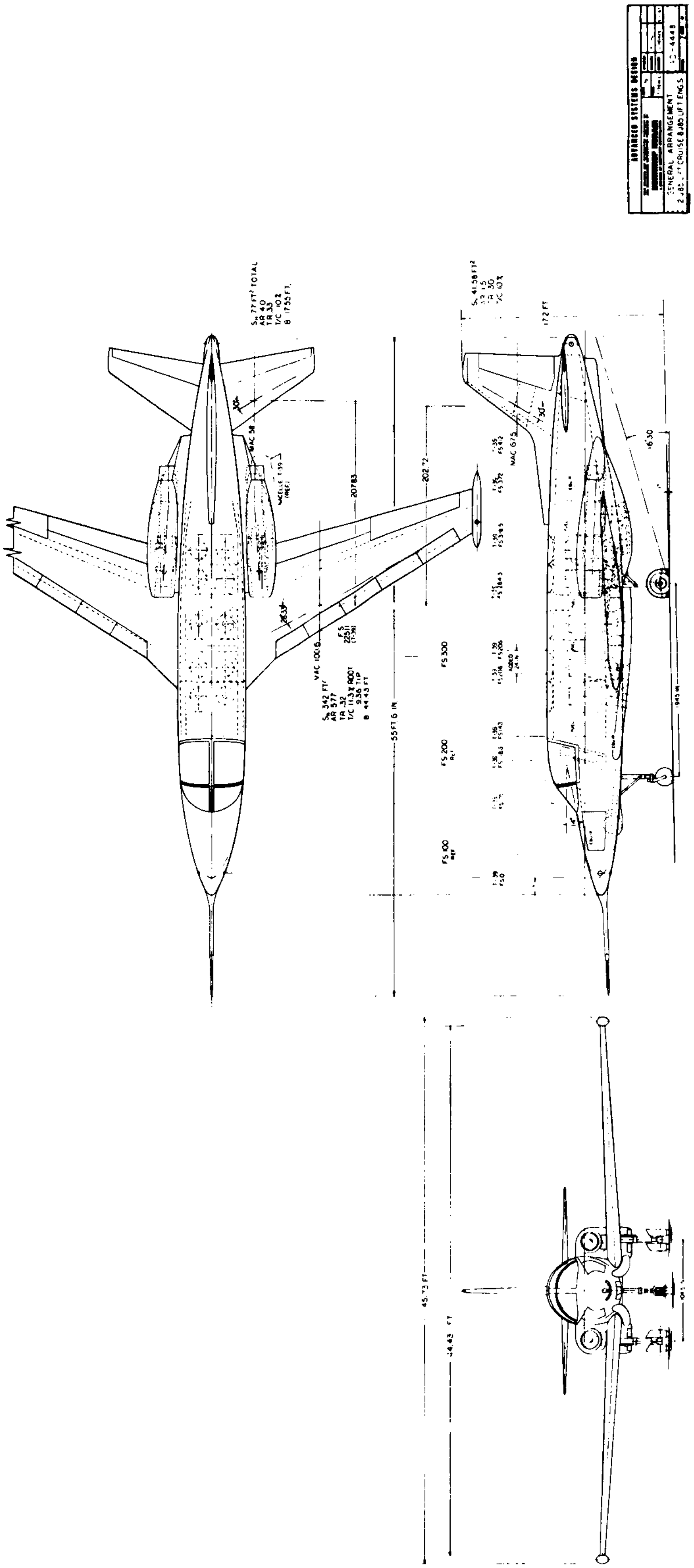


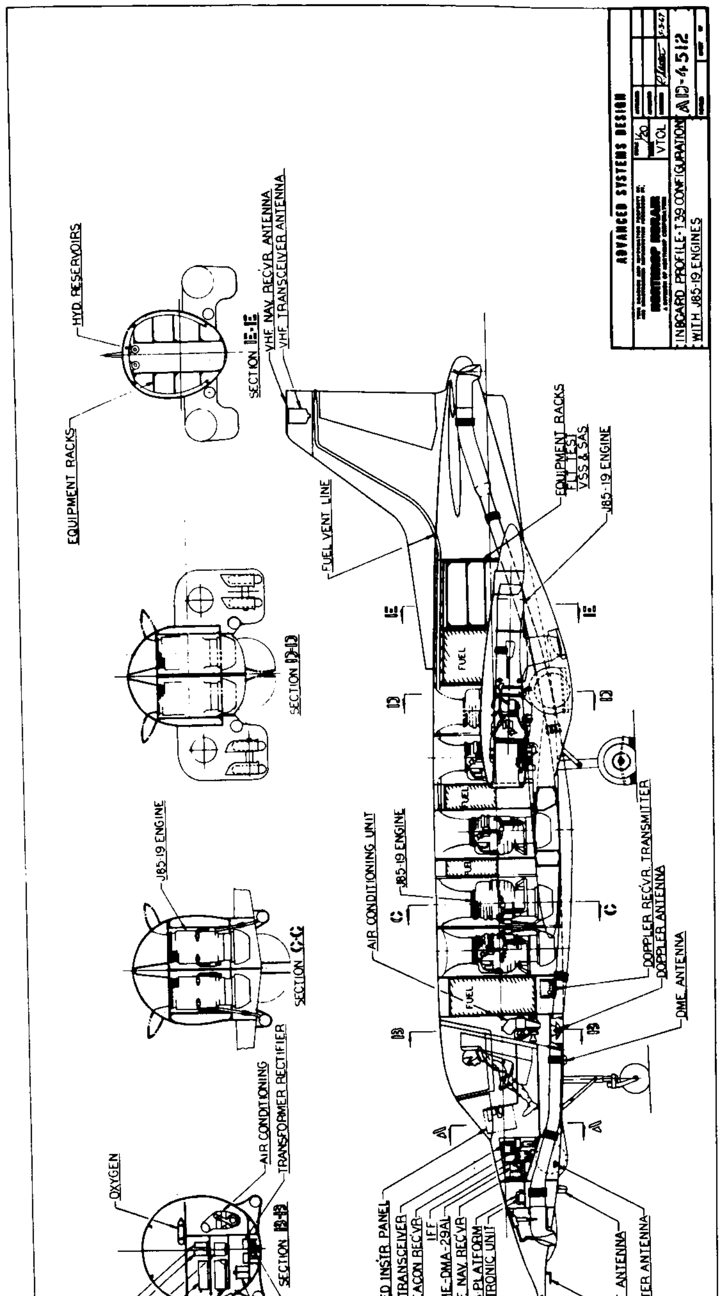
COMPOSITE
DIRECT LIFT
COMPOSITE
DIRECT LIFT
COMPOSITE
DIRECT LIFT
60
58
56
54
52
50
48
46
44
42
40
38
18
16
14
MOMENT OF INERTIA ~ SLUG-FEET^2
21
20
19
18
17
16
15
14
13
GROSS WEIGHT ~ 1000 POUNDS

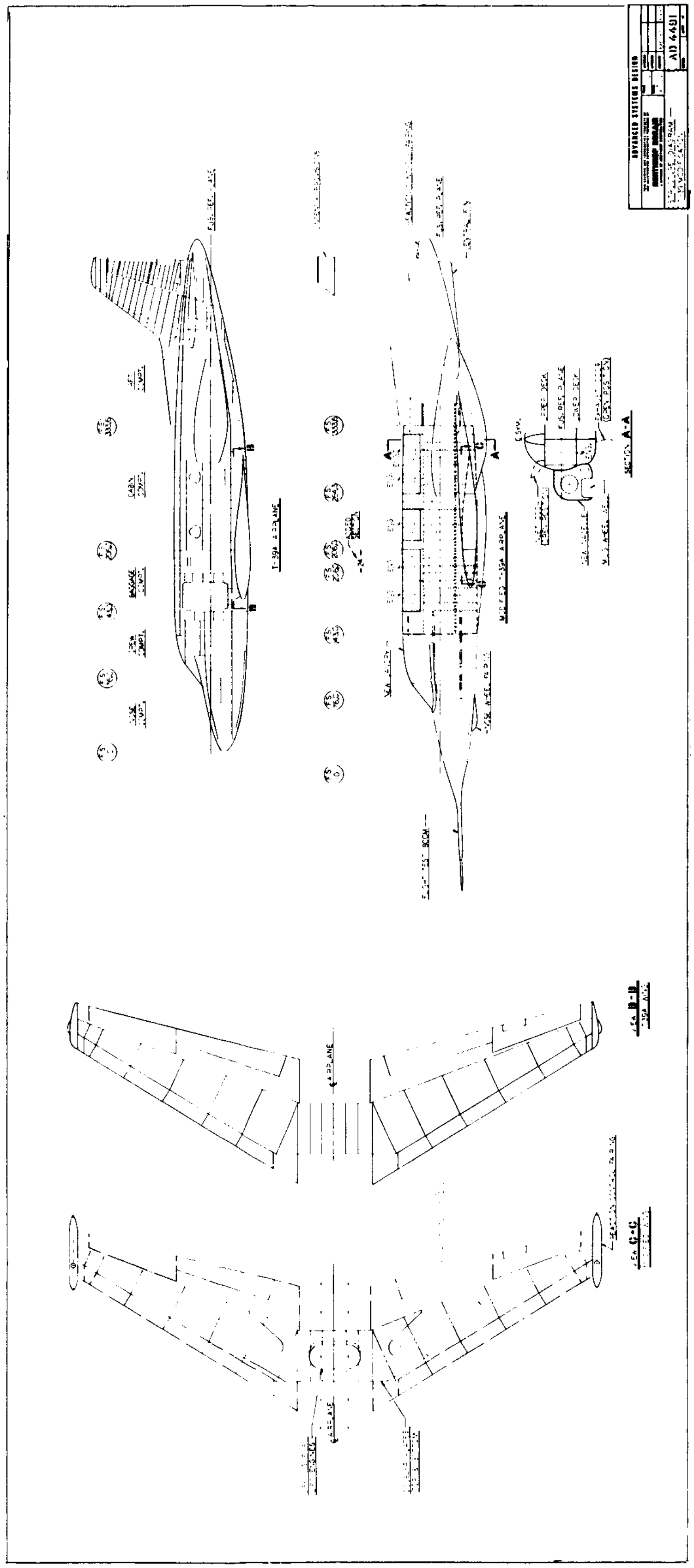


Operating Weight Empty 15,166 lbs.
Usable Fuel 4,234 lbs.
18.0
17.0
16.0
15.0
14.0
13.0
12.0
11.0
10.0
9.0
8.0
7.0
6.0
5.0
4.0
HOVER TIME - MINUTES
AD-4448A
-10
-5
0
+5
+10
PERCENT CHANGE - OPERATING WEIGHT EMPTY

AN-91030 Group Weight Statement	Accuracy Analysis 2%	Other Weight 5%	Accountable Weight	AD-4448 Estimated Weight
Wing Group		406	1,389	1,795
Tail Group			311	311
Body Group		1,731	824	2,555
Alighting Gear Group		297	792	1,089
Surface Controls Group		508		508
Engine Section			891	891
Propulsion Group		902	4,294	5,196
Instrument Group		171		171
Hydraulic Group		190		190
Electrical Group		335		335
Electronics Group		61	159	220
Furn. & Equip. Group		410		410
Air Cond. Group		68		68
Hovering Controls		625		625
Weight Empty		5,704	8,660	14,364
Useful Load				
Crew			400	400
Engine Oil			70	70
Unusable Fuel		45		45
Operating Weight Empty		5,749	9,130	14,879
Contingency		287		287


ADVANCED SYSTEMS DESIGN
GENERAL ARRANGEMENT
55 FT 6 IN
FS 100
FS 200
FS 300


ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
VTOL
INBOARD PROFILE-T39 CONFIGURATION WITH J85-19 ENGINES
AD-4512
HYD RESERVOIRS
EQUIPMENT RACKS
SECTION E-E
VHF NAV RECVR ANTENNA
VHF TRANSCEIVER ANTENNA
FUEL VENT LINE
EQUIPMENT RACKS
VSS & SAS
J85-19 ENGINE
SECTION D-D
J85-19 ENGINE
SECTION C-C
AIR CONDITIONING UNIT
J85-19 ENGINE
FUEL
DOPPLER RECVR TRANSMITTER
DOPPLER ANTENNA
DME ANTENNA
OXYGEN
AIR CONDITIONING
TRANSFORMER RECTIFIER
SECTION B-B
RADAR ALTIMETER
ANGULAR 3 AXIS ACCEL
3 AXIS ACCEL
ELECTRONIC ASSEM UNIT
VECTOR CONTROL POWER SERVO AMP
BATTERY
DOPPLER ANTENNA
SECTION A-A
INTEGRATED INSTR PANEL
RTA-41B VHF TRANSCEIVER
MKA-28 MARKER BEACON RECVR
IFF
DME-DMA-29AL
RNA-26C VHF NAV RECVR
SR-3 PLATFORM
ELECTRONIC UNIT
VSS SENSORS
MARKER BEACON ANTENNA
IFF ANTENNA
RADAR ALTIMETER ANTENNA



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6.0 SUBSYSTEMS

6.1 PRIMARY FLIGHT CONTROLS

6.1.1 Control System Routing

Drawing AD 4498 (Figure 6-1) shows the tentative control mechanization and its location and the cable routing.

6.1.2 Surface And Jet Nozzle Actuation

The three rotational axes (pitch, yaw and roll) control systems will be powered by dual hydraulic systems. Failure of either system will result in reduced hinge moment available but will not result in any transients unless the control surface at the time of hydraulic failure is loaded higher than the hydraulic relief pressure of the remaining system.

Each aileron will be powered by both hydraulic systems, and the cable control system across the airplane will be fitted with preloaded springs. Failure of one actuator or the control system for one wing will result in reduced control power but will not restrict motion of the opposite aileron. Failure of one hydraulic system will result in reduced control power.

The reaction jet nozzle at each wing tip will be driven by the same actuator that moves the aerodynamic surface. The scheduling will be modified by linkage, and series SAS actuators will be included in the mechanization. SAS will, by this approach be effective on the reaction system only on the roll axis. Electric actuators will be included so that the gain of the reaction system can be varied and so that the nozzles can be closed for duct pressurization and during cruise flight.

The pitch control actuation system is somewhat similar to that for the roll axis, but the transition trim requirements and the lack of a surface actuator in the nose of the airplane dictates minor differences. The yaw actuation system will be similar to the pitch mechanization except that there is no need for aerodynamic series trim.

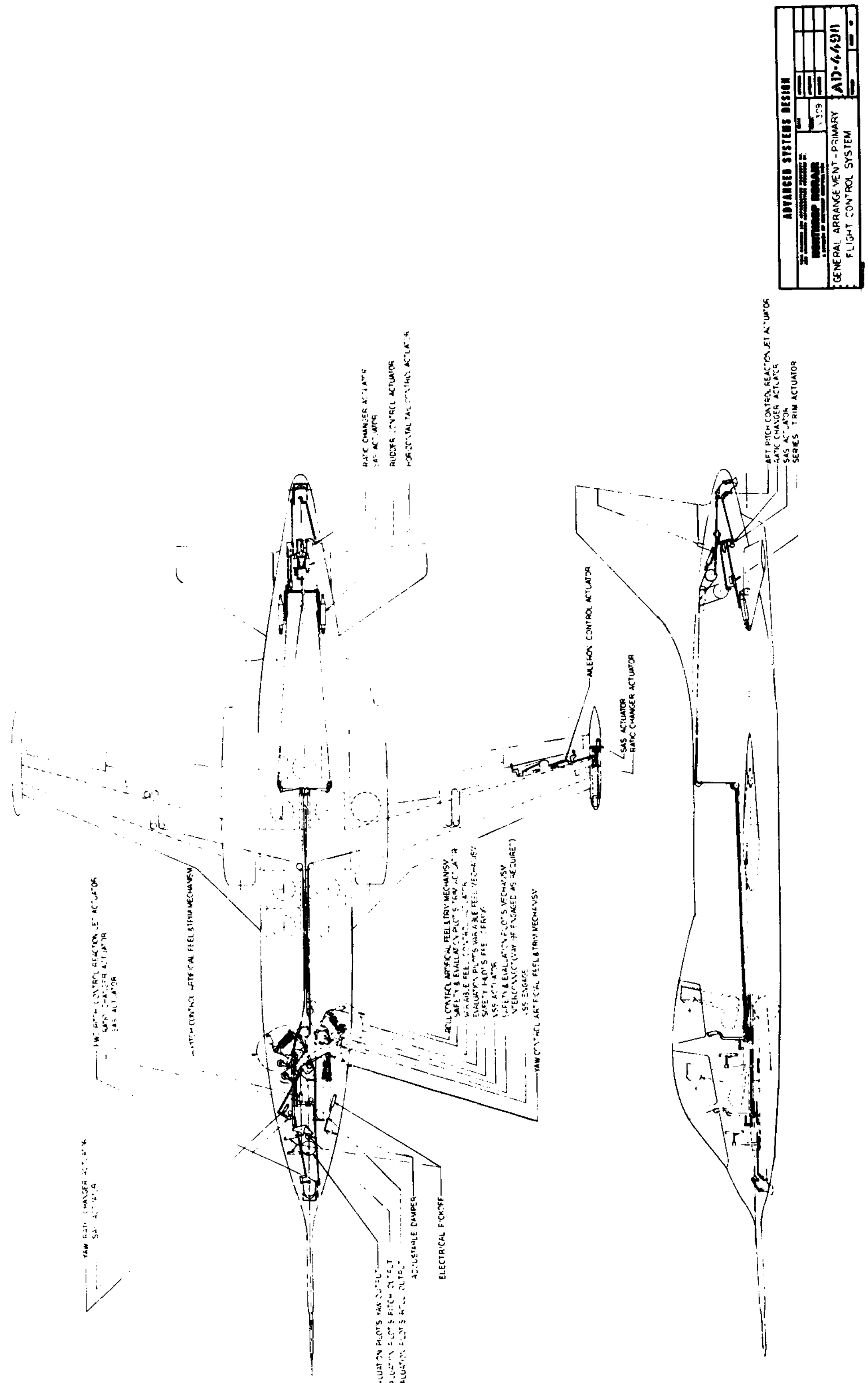

ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
GENERAL ARRANGEMENT - PRIMARY
FLIGHT CONTROL SYSTEM
AILERON CONTROL ACTUATOR
SAS ACTUATOR
RATIO CHANGER ACTUATOR
ADJUSTABLE DAMPER
ELECTRICAL PICKOFF
SERIES TRIM ACTUATOR

FIGURE 6-1

It is expected that off-the-shelf actuators can be used. The T-38/F-5 rudder actuators may be used both on the rudder and on the aileron systems. The X-21 aileron actuators are dual tandem, and will be adequate for the horizontal tail on the NASA/VTOL research airplane. The horizontal tail travel required for transition trim may pose a sensitivity matching and safety problem. Ratio changing and auxiliary pitch trim control, or a trimmable tail and elevator system may be required. Drawing AD-4518 shows block diagrams of all three primary control systems.

6.1.3 Artificial Feel Mechanization

To avoid severe control transients when the control is transferred from one control station to the other it is desirable that both the evaluation pilots and the safety pilots controls be synchronized at any no-load trim point. The arrangement shown in drawing AD-4518 (Figure 6-2) provides trim synchronization. The feel forces felt by either pilot is provided by springs, with trim accomplished by changing the structural reference point.

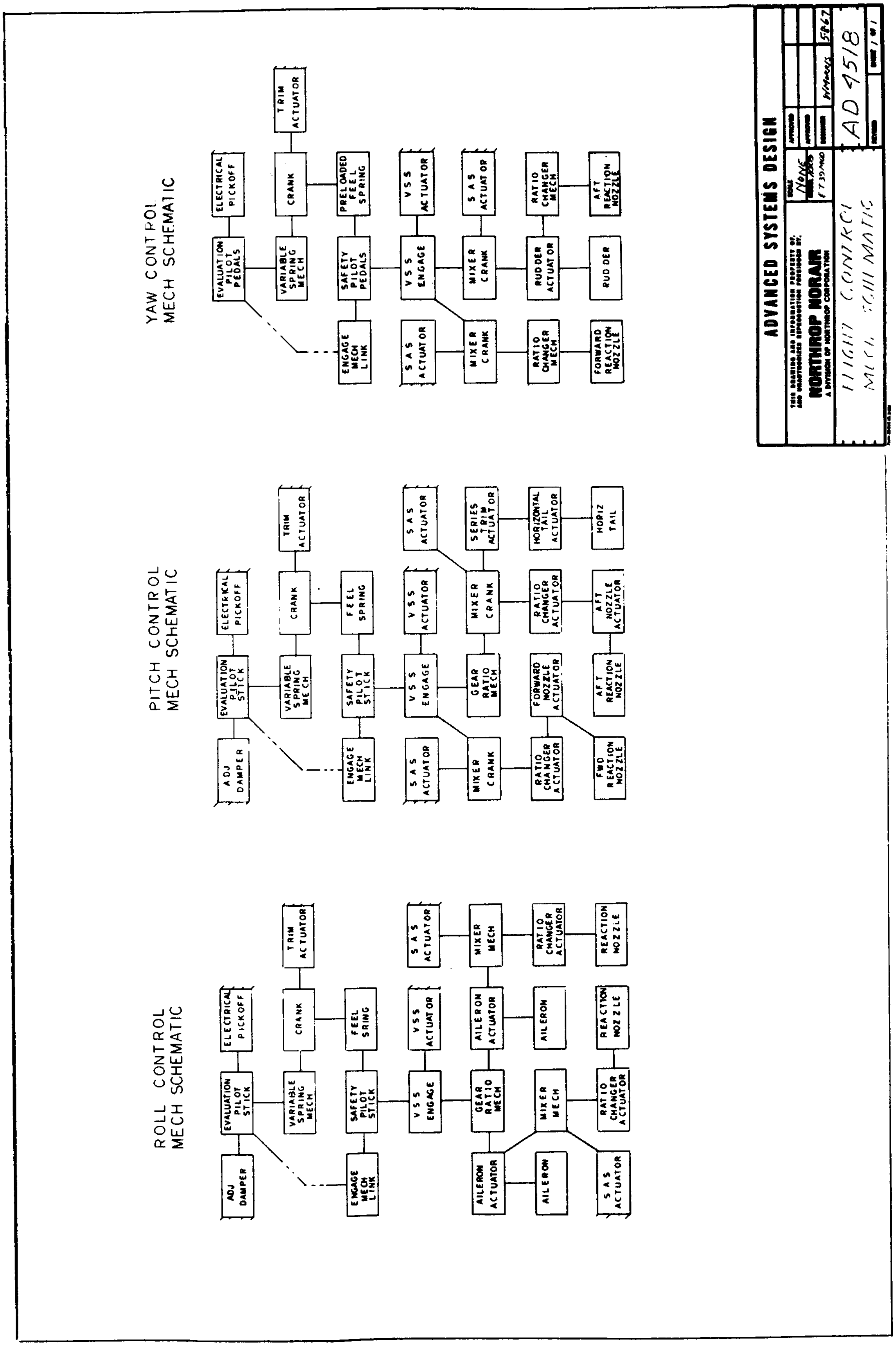
The feel spring for the evaluation pilot can be varied in flight to provide a smooth change of gradient, and the trim reference can also be changed as required or as scheduled. The feel spring for the safety pilot can be trimmed, but does not provide in-flight variation of gradient. This obviously could be provided if desired.

The damper shown on the evaluation pilots control mechanization is ground adjustable.

6.2 WING FLAP SYSTEM

The wing flap system consists of trailing edge and leading edge flaps, electrically powered and operated through a three-position flap lever. See Wing Flaps Configuration drawing (Figure 6-3). Full flap extension or retraction (leading and trailing edge flaps) will take 10 to 17 seconds. Two AC motors, one driven by one electrical source the other by the opposite electrical system, operate each set of flaps through gear reduction units and actuators. The two trailing edge flaps are mechanically interconnected by flexible rotary shafting to ensure flap operation should one motor or actuator fail.

Flap operation will be at a slower than normal rate if one motor or actuator fails. The two leading edge flaps are mechanically interconnected in the same manner.



YAW CONTROL MECH SCHEMATIC
ELECTRICAL PICKOFF
TRIM ACTUATOR
EVALUATION PILOT PEDALS
CRANK
VARIABLE SPRING MECH
PRELOADED FEEL SPRING
SAFETY PILOT PEDALS
ENGAGE MECH LINK
VSS ACTUATOR
VSS ENGAGE
SAS ACTUATOR
SAS ACTUATOR
MIXER CRANK
MIXER CRANK
RATIO CHANGER MECH
RUDDER ACTUATOR
RATIO CHANGER MECH
AFT REACTION NOZZLE
RUDDER
FORWARD REACTION NOZZLE
PITCH CONTROL MECH SCHEMATIC
TRIM ACTUATOR
ELECTRICAL PICKOFF
CRANK
EVALUATION PILOT STICK
ADJ DAMPER
VARIABLE SPRING MECH
FEEL SPRING
SAFETY PILOT STICK
ENGAGE MECH LINK
SAS ACTUATOR
VSS ACTUATOR
VSS ENGAGE
SAS ACTUATOR
SERIES TRIM ACTUATOR
MIXER CRANK
GEAR RATIO MECH
MIXER CRANK
HORIZONTAL TAIL ACTUATOR
RATIO CHANGER ACTUATOR
FORWARD NOZZLE ACTUATOR
RATIO CHANGER ACTUATOR
HORIZ TAIL
AFT NOZZLE ACTUATOR
AFT REACTION NOZZLE
FWD REACTION NOZZLE
ROLL CONTROL MECH SCHEMATIC
TRIM ACTUATOR
ELECTRICAL PICKOFF
CRANK
EVALUATION PILOT STICK
ADJ DAMPER
VARIABLE SPRING MECH
FEEL SPRING
SAFETY PILOT STICK
ENGAGE MECH LINK
SAS ACTUATOR
VSS ACTUATOR
VSS ENGAGE
MIXER MECH
AILERON ACTUATOR
GEAR RATIO MECH
AILERON ACTUATOR
RATIO CHANGER ACTUATOR
AILERON
MIXER MECH
AILERON
REACTION NOZZLE
REACTION NOZZLE
RATIO CHANGER ACTUATOR
SAS ACTUATOR
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
A DIVISION OF NORTHROP CORPORATION
NONE
AD 4518

FIGURE 6-2

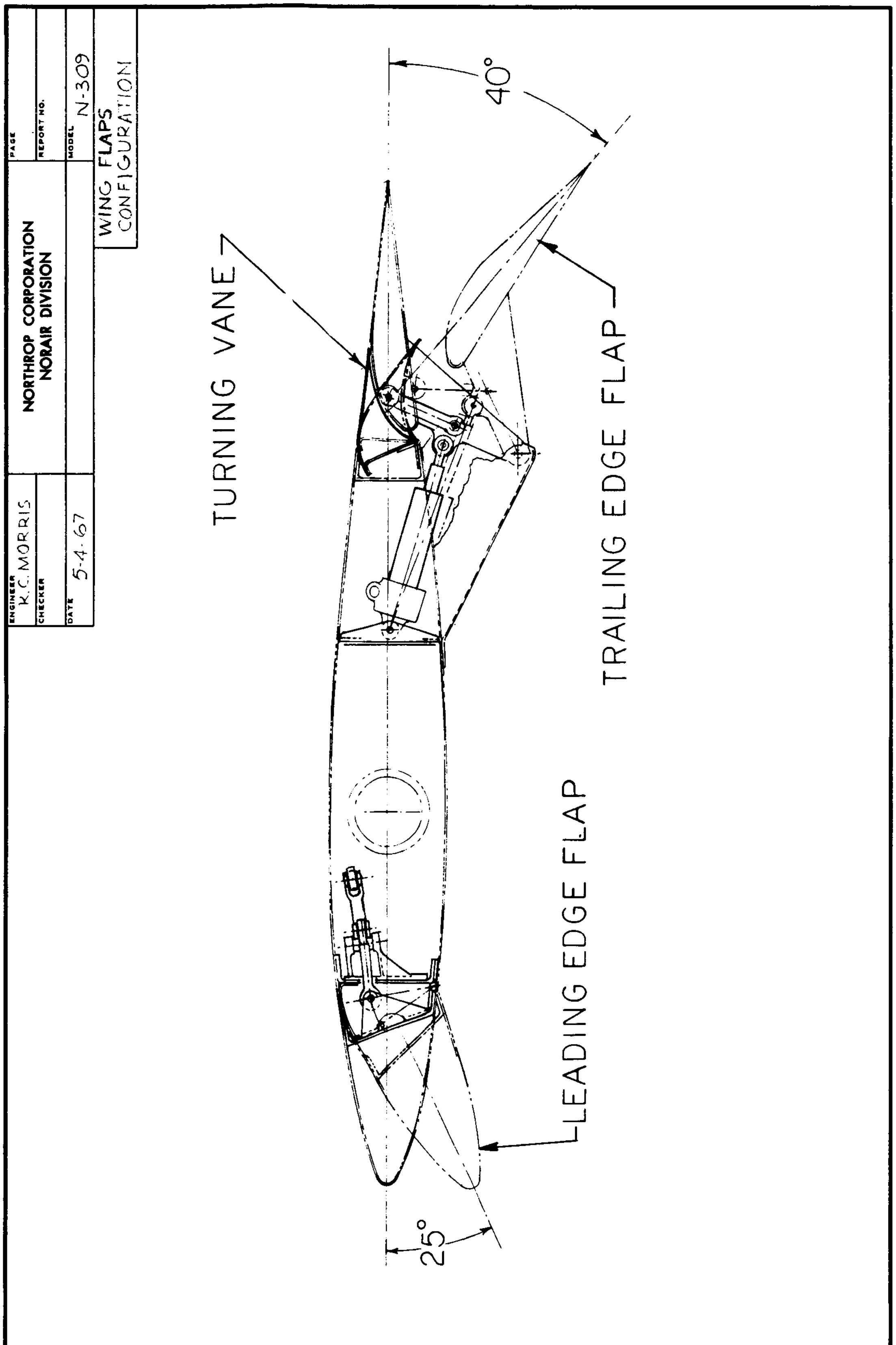

NORTHROP CORPORATION
NORAIR DIVISION
PAGE
REPORT NO.
MODEL N-309
WING FLAPS
CONFIGURATION
ENGINEER K.C. MORRIS
CHECKER
DATE 5-4-67
TURNING VANE
TRAILING EDGE FLAP
LEADING EDGE FLAP
40°
25°

FIGURE 6-3

6.2.1 Wing Flap Lever And Flap Position Indicator

The wing flap lever is a three-position lever controlling the electrical operation of the leading and trailing edge flaps. The placarded positions are "up", "mid" and "full". These positions are visually displayed to the pilot by the flap position indicator on the instrument panel. With the lever in the "mid" position, the leading edge flaps are fully extended and the trailing edge flaps are at approach position. Placing the lever in the "full" position extends both leading and trailing edge flaps to full extension.

6.3 HYDRAULIC SYSTEMS

Two hydraulic systems will be used for attitude control as well as the landing gear retraction system, nose wheel steering, engine doors and thrust diverters. Emergency wheel brakes are provided by a master cylinder in the brake valve. Landing flaps are operated by electrical power, and manual release of the landing gear will allow it to fall free to the down and locked position with the aid of bungee cylinders. Components and system design technology will be completely within the "state-of-the-art", making development programs unnecessary.

Hydraulic system heat generated for the most part during hover flight will be removed from the system to restrict oil temperature to 275°, allowing the use of proven and reliable components.

Reliability requirements will be established for the components and systems using information gained on previous projects. The V/STOL vehicle then will have the advantage of components which are designed for usage far in excess of the flight program for this vehicle. All four hydraulic pumps are driven by the lift/cruise engines. Figure 6-4 shows a block diagram of the system. Drawing AD 4496 (Figure 6-5) is the complete hydraulic system schematic. The primary and utility controls are each powered by two hydraulic pumps. One pump is attached to the accessory drive gear box. The other pump is mounted on the starter pad of the opposite engine. The loss of either engine will not cause the loss of a complete system and requires no pilot effort to switch to an emergency system.

The estimated flow requirements indicated that, as far as the hydraulic system is concerned, either a vertical or normal landing can be accomplished with either cruise engine out. Windmilling power may be adequate for a conventional landing even if both cruise engines are out. Table 6-1 shows maximum flow requirements for each function and for each system.

TABLE 6-1 ESTIMATED HYDRAULIC POWER REQUIREMENTS

FUNCTION	NO. 1 SYSTEM REQUIREMENT		NO. 2 SYSTEM REQUIREMENT	
	MAX. FLOW REQUIREMENT	AVERAGE REQUIREMENT	MAX. FLOW REQUIREMENT	AVERAGE REQUIREMENT
AILERON	3.52 GPM	1.172 GPM	3.52 GPM	1.172
RUDDER	0.58 GPM	0.193 GPM	0.58 GPM	0.193
HORIZONTAL TAIL	6.00 GPM	2.00 GPM	6.00 GPM	0.40
				2.00
NOSE WHEEL STEERING			0.30 GPM	0.10
LANDING GEAR			4.00 GPM	1.33
CRUISE ENGINE THRUST DIVERTOR			1.94 GPM	0.647
LIFT ENGINE UPPER AND LOWER DOORS			2.00 GPM	0.666
JET REACTION VALVES			5.40 GPM	1.800
STABILITY AUGMENTATION SYSTEM	0.40 GPM	0.40 GPM	0.40 GPM	0.40
VARIABLE STABILITY SYSTEM	0.36 GPM	0.36 GPM		
TOTAL	10.86 GPM	4.125 GPM	24.14 GPM	8.593 GPM

TOTAL POWER AVAILABLE
EQUAL TO A MINIMUM OF 20 GPM

Maximum flow requirement is equivalent to maximum surface rate at maximum hinge moment.
Average flow requirement is equivalent to 1 second maximum flow and hinge moment followed by 2 seconds quiescent flow.

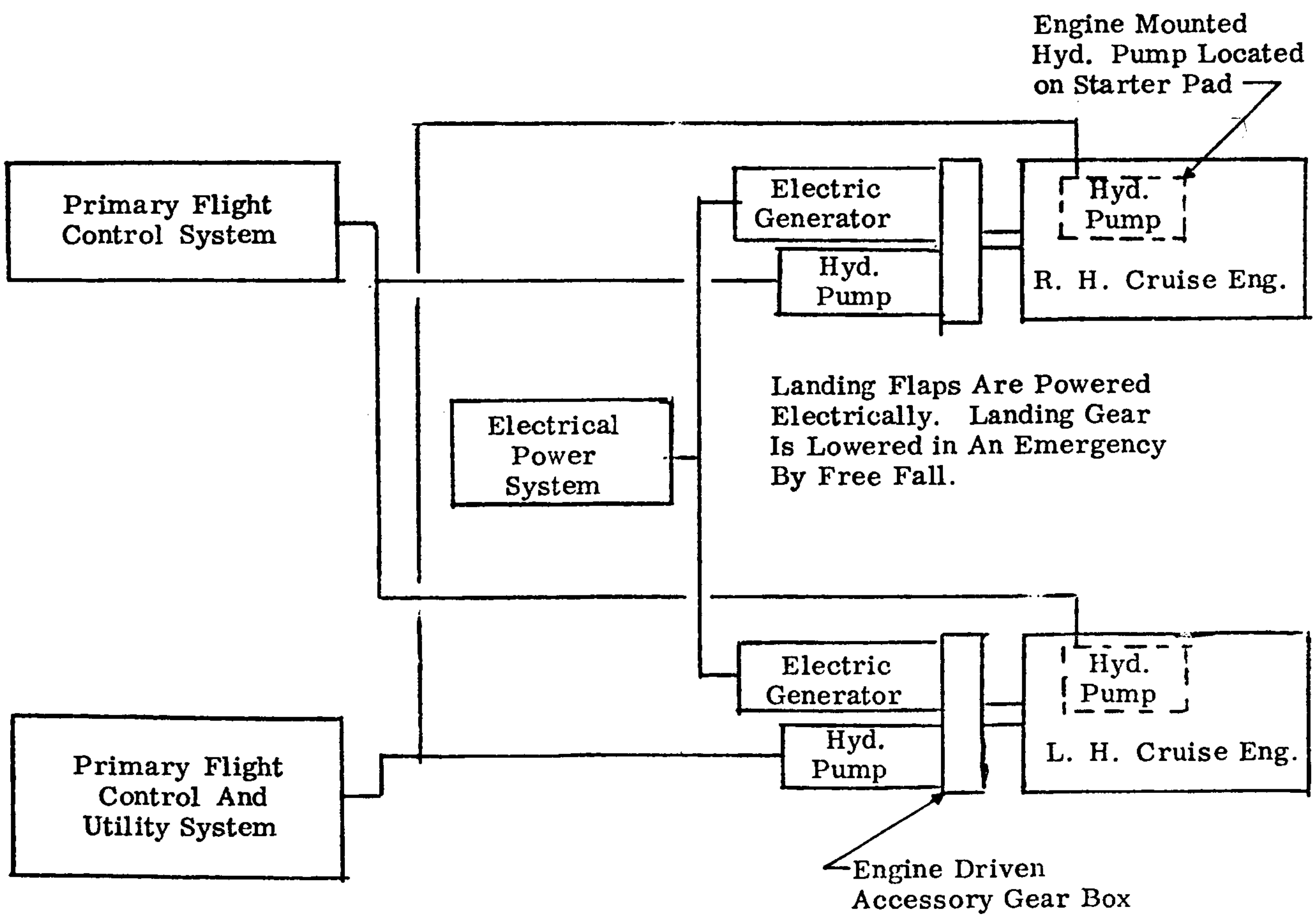


FIGURE 6-4 HYDRAULIC SYSTEM BLOCK DIAGRAM

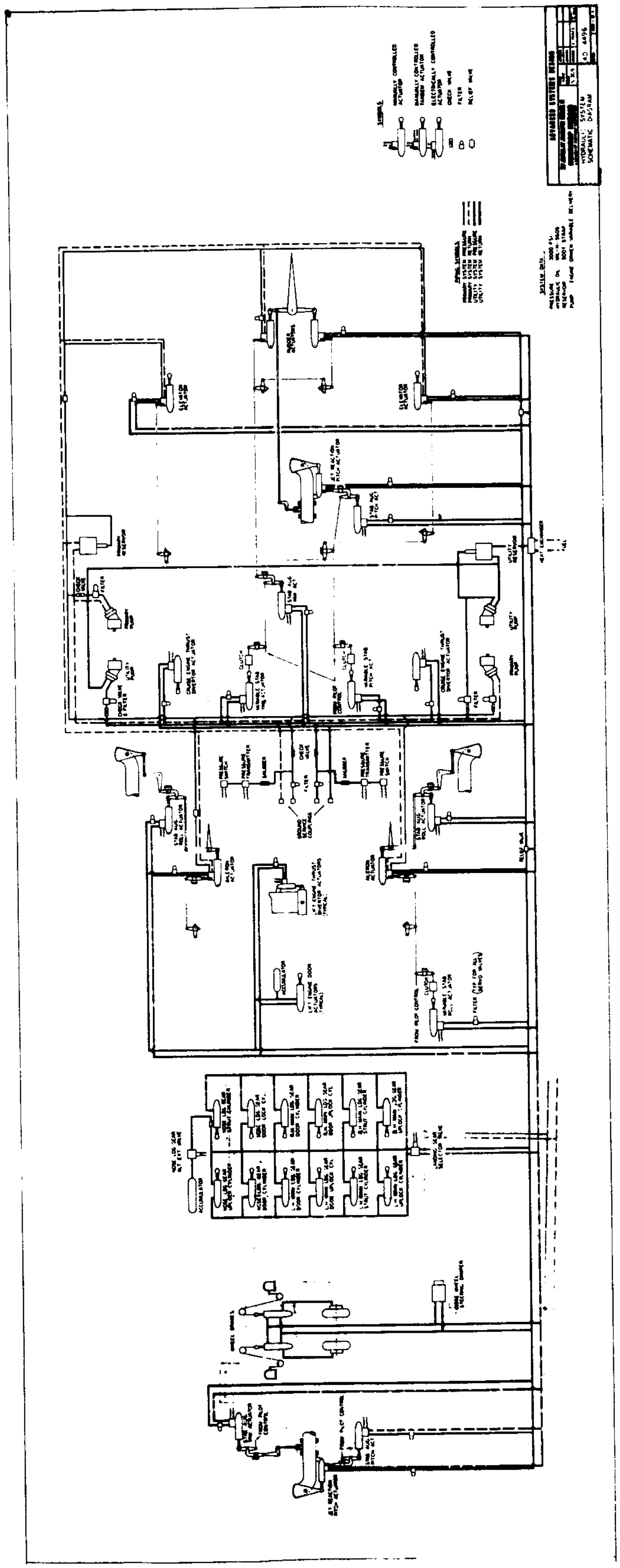


FIGURE 6-5

6.4 FUEL SYSTEM - N309

The normal fuel load of 4300 pounds of which 300 pounds is used for engine start and pre-flight warmup, is carried in six bladder cells, arranged in three LH and RH pairs of fuselage tanks as shown in drawing AD 4519. The center pair, auxiliary tanks, drain by gravity into the forward and aft pairs of main tanks which hold a majority of the fuel. An additional 300 pounds may be carried in the aft main tanks for STOL missions or additional warmup fuel. Each pair of main tanks is interconnected to provide added reliability, and inverted flight capability can be easily provided for inadvertent emergency operation. Four boost pumps, one in each main tank, supply fuel to all engines, but any three pumps can supply critical hover fuel flows. In case of a single boost pump failure, a cockpit warning light would advise the pilot, so that he may turn off one of the two pumps in the opposite pair of main tanks before excessive CG shift is encountered. Inasmuch as engine specifications require operation with aircraft boost pumps inoperative at altitude conditions more stringent than those expected in hover operation of this aircraft, a pump failure, or even complete electrical failure could be experienced without causing a critical landing emergency. Two gravity fillers, one in each auxiliary tank, provide for complete fueling of the airplane through the tank interconnects. All tanks are vented to a single, open vent line exhausting from the vertical stabilizer.

6.5 FUEL SYSTEM - T-39A MODIFIED

The fuel system for the modified T-39A is basically similar to that shown for the N-309 airplane. Four auxiliary tanks are used in place of the two shown on drawing AD 4519 (Figure 6-6). The arrangement may be seen on drawing AD 4512, T-39A Inboard Profile (Figure 6-7).

6.6 MODEL N-309 LANDING GEAR GENERAL DESCRIPTIONS

The landing gear arrangement as shown by Drawing AD 4495 (Figure 6-8) is of the tricycle type utilizing hydropneumatic shock struts. The nose and main gears are hydraulically actuated, and the struts are mechanically compressed during retraction. The gear doors are hydraulically actuated and a positive locking system is provided for both the gear extended and retracted positions. In addition, both the nose and main landing gear shock struts are equipped with a shock strut pressure sensing device and instrumented in the cockpit to permit pilot checkout of the reaction controls prior to takeoff.

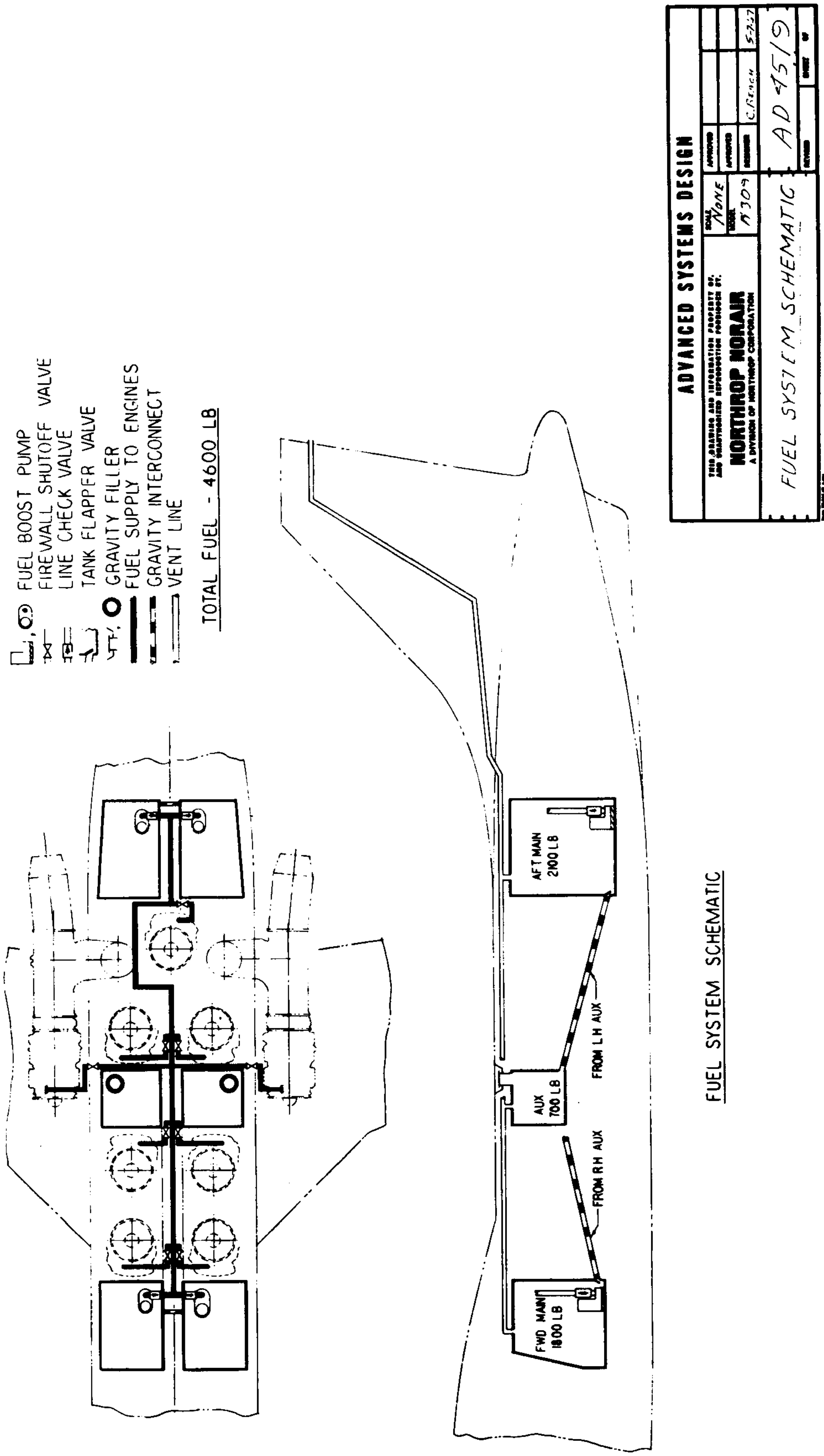

FUEL BOOST PUMP
FIREWALL SHUTOFF VALVE
LINE CHECK VALVE
TANK FLAPPER VALVE
GRAVITY FILLER
FUEL SUPPLY TO ENGINES
GRAVITY INTERCONNECT
VENT LINE
TOTAL FUEL - 4600 LB
FWD MAIN 1800 LB
AUX 700 LB
AFT MAIN 2100 LB
FROM RH AUX
FROM LH AUX
FUEL SYSTEM SCHEMATIC
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
A DIVISION OF NORTHROP CORPORATION
NONE
N309
C. REACH
5-7-57
FUEL SYSTEM SCHEMATIC
AD 45/9

FIGURE 6-6

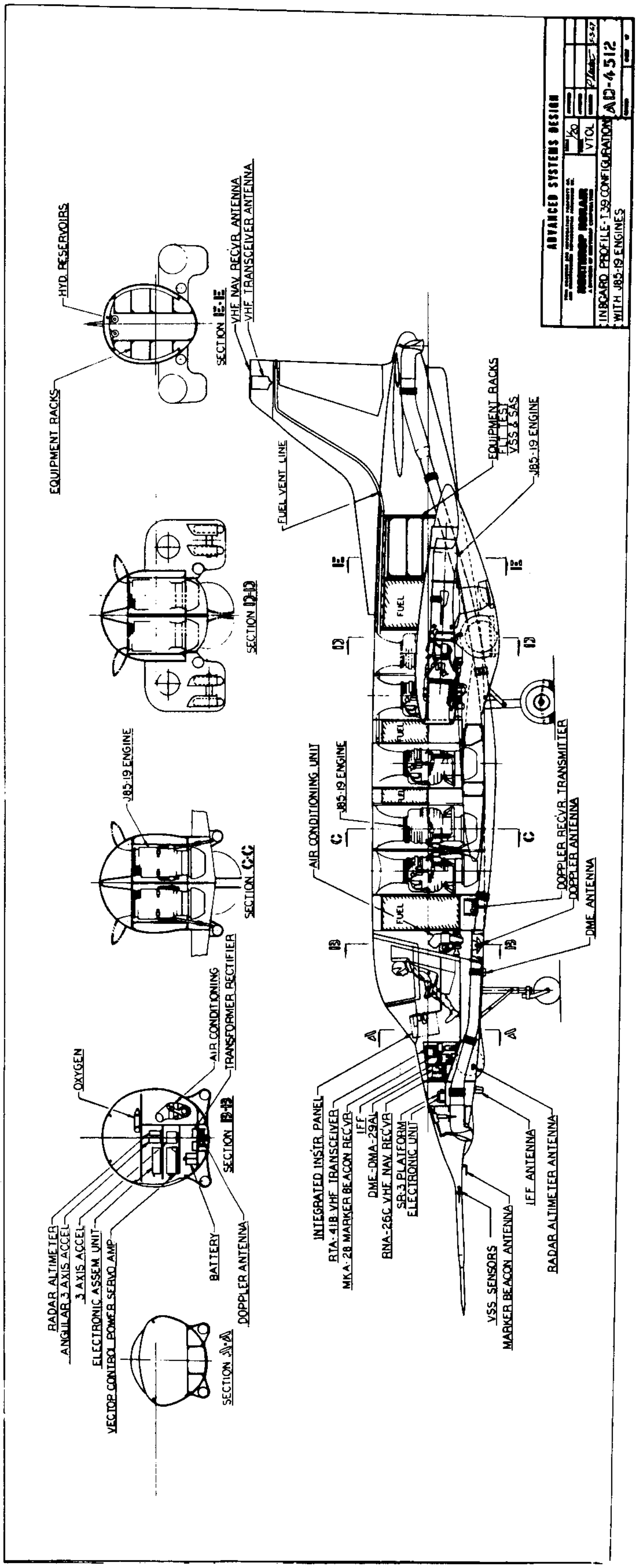

ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
INBOARD PROFILE-T-39 CONFIGURATION WITH J85-19 ENGINES
VTOL
AD-4512
HYD RESERVOIRS
EQUIPMENT RACKS
SECTION E-E
VHF NAV RECVR ANTENNA
VHF TRANSCEIVER ANTENNA
FUEL VENT LINE
EQUIPMENT RACKS
FLT TEST
VSS & SAS
J85-19 ENGINE
SECTION D-D
J85-19 ENGINE
SECTION C-C
AIR CONDITIONING UNIT
J85-19 ENGINE
FUEL
DOPPLER RECVR TRANSMITTER
DOPPLER ANTENNA
DME ANTENNA
OXYGEN
AIR CONDITIONING
TRANSFORMER RECTIFIER
SECTION B-B
RADAR ALTIMETER
ANGULAR 3 AXIS ACCEL
3 AXIS ACCEL
ELECTRONIC ASSEM UNIT
VECTOR CONTROL POWER SERVO AMP
BATTERY
DOPPLER ANTENNA
SECTION A-A
INTEGRATED INSTR PANEL
RTA-41B VHF TRANSCEIVER
MKA-28 MARKER BEACON RECVR
IFF
DME-DMA-29AL
RNA-26C VHF NAV RECVR
SR-3 PLATFORM
ELECTRONIC UNIT
VSS SENSORS
MARKER BEACON ANTENNA
IFF ANTENNA
RADAR ALTIMETER ANTENNA

FIGURE 6-7

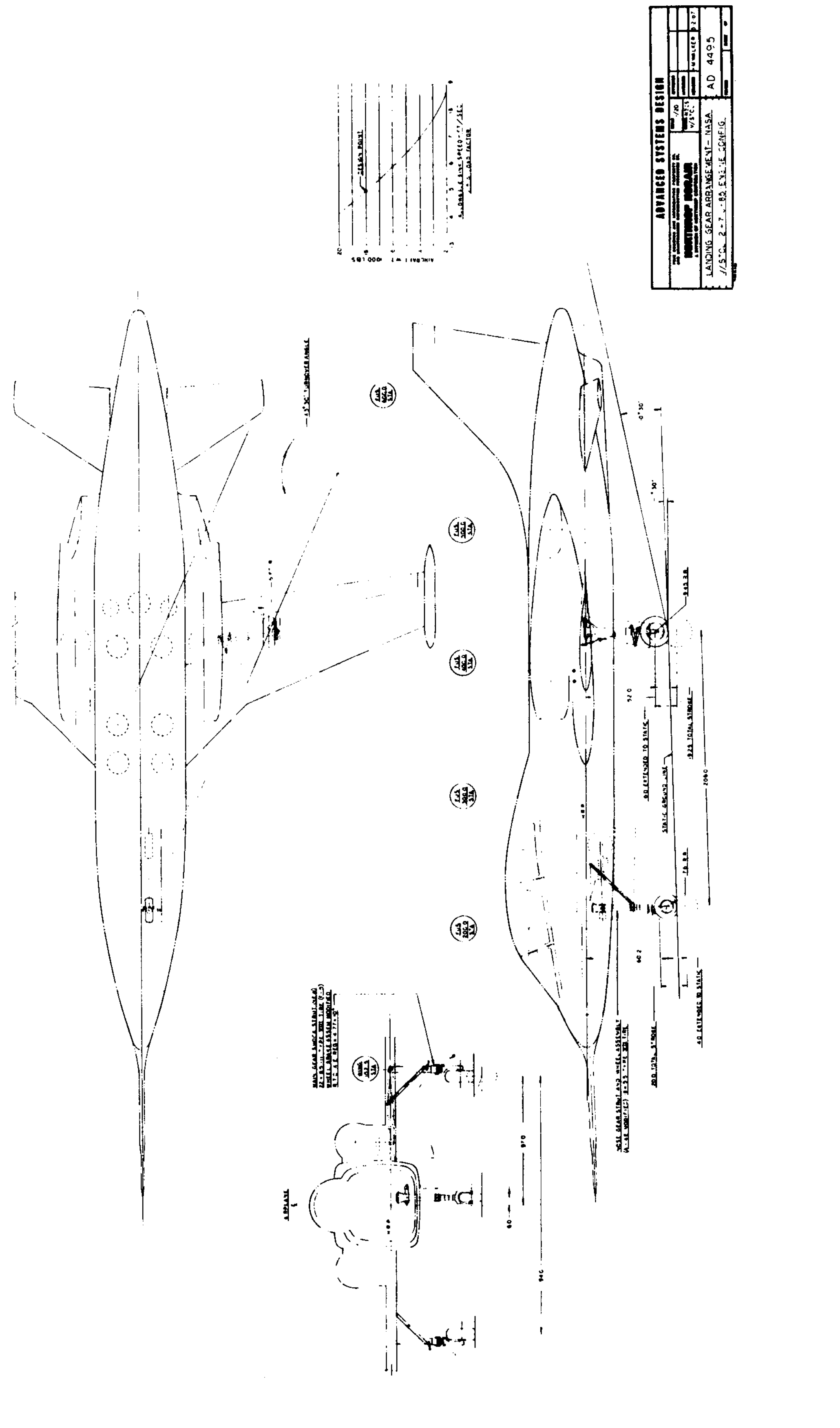

ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
LANDING GEAR ARRANGEMENT - NASA
AD 4495

FIGURE 6-8

6.6.1 Nose Landing Gear

The nose landing gear utilizes a modified A-4E shock strut tire and wheel assembly. The A-4E nose landing gear shock strut requires modification of the metering pin for landing kinetic energy requirements, plus some minor structural modification for attachment of a new trunnion attach fitting. The geometry and installation arrangement are shown on Drawing AD 4494, Figure 6-9.

The nose gear shock strut is trunnion-mounted in the forward fuselage at fuselage station 215.5, six inches to the left of the airplane center line and retracts aft and up into the fuselage. The shock strut is mechanically compressed during retraction. The gear is secured in the down position by a foldable type drag brace and an over-center type lock linkage. The gear is hydraulically actuated and secured in the up position by positive type locks. The nose gear is fully faired in the retracted position by two doors, a strut door and a wheel door. The strut door is mechanically linked to the strut and is actuated with the strut; the wheel door is hydraulically actuated and returns to closed position after gear extension. The A-4E nose wheel is normally equipped with a hydraulic shimmy damper only; however, new or later A-4 aircraft are presently being equipped with nose wheel steering units and it is therefore assumed it could be made available for this installation.

6.6.2 Main Landing Gear

The main landing gear employs a new shock strut and utilizes the F-5 (22 x 8.5 - 11.0 Type VIII) tire, wheel and modified brake assembly. The brake assembly requires modification for reduction of kinetic brake energy and the resultant weight savings. The main landing gear geometry and installation is shown by Drawing AD 4493 (Figure 6-10). The strut is trunnion mounted and attaches to the wing structure at wing station 102.5 and is laterally stabilized by a folding type side brace. The main gear is hydraulically actuated and retracts inboard and is housed within the wing and engine pod structural fairing. The shock strut is mechanically compressed during retraction for stowage, and secured in the up position by combined mechanical and hydraulically actuated locks. The main gear is secured in the extended position by a mechanical over center type linkage attached to structure and the upper half of the side brace assembly. Each main gear bay is fully faired after gear retraction by two doors. The wheel door is hydraulically actuated and returns to the closed position after gear extension. The other or strut door is mechanically linked to the strut and is actuated by gear motion.

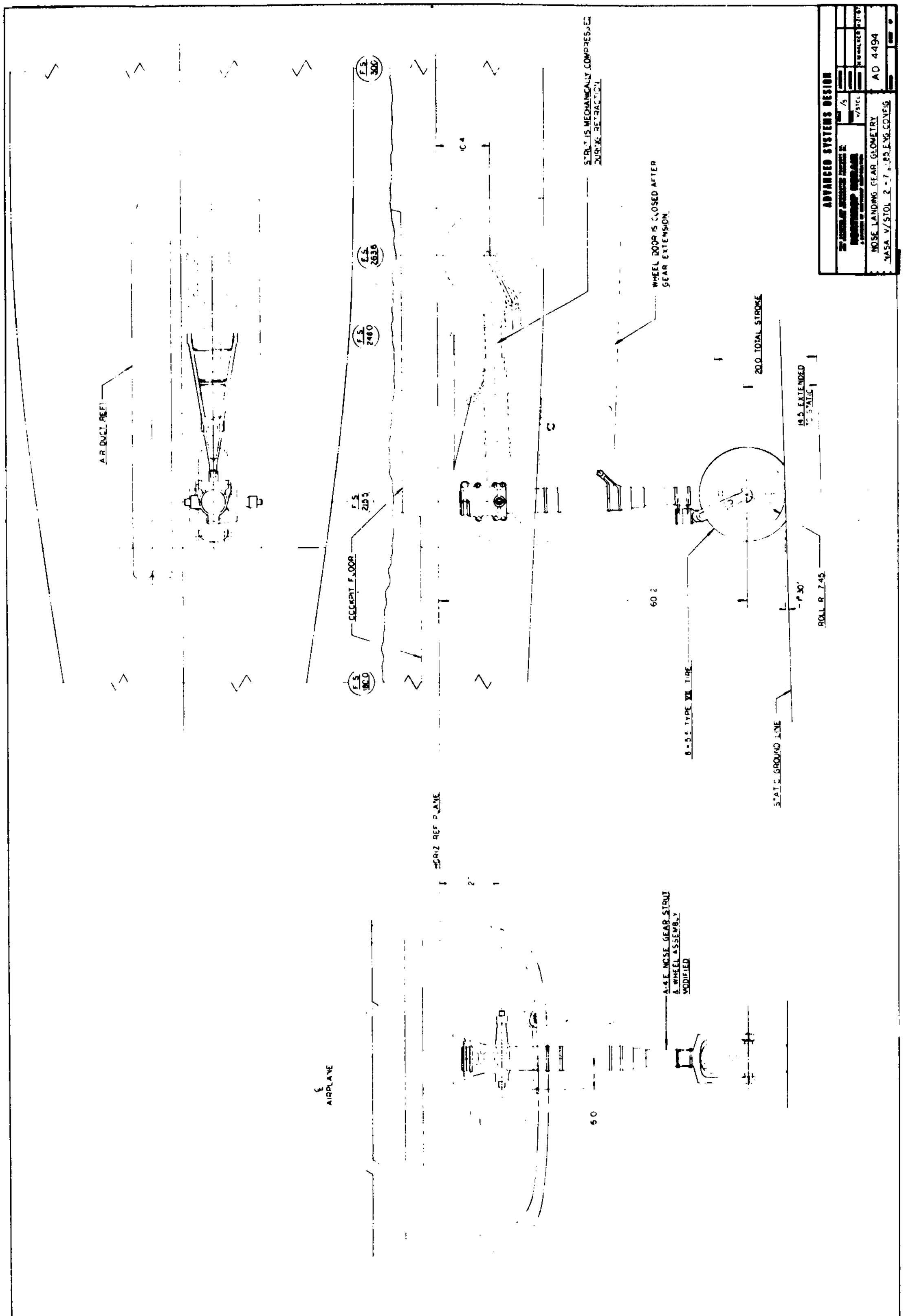

ADVANCED SYSTEMS DESIGN
NOSE LANDING GEAR GEOMETRY
NASA V/STOL 2+7+.85 EVG CONFIG
AD 4494
STRUT IS MECHANICALLY COMPRESSED DURING RETRACTION
WHEEL DOOR IS CLOSED AFTER GEAR EXTENSION
2.00 TOTAL STROKE
14.5 EXTENDED
ROLL R 7.45
STATIC GROUND LINE
6 x 5.5 TYPE VII TIRE
COCKPIT FLOOR
AIR DUCT REF
HORIZ REF PLANE
A-4E NOSE GEAR STRUT & WHEEL ASSEMBLY MODIFIED
℄ AIRPLANE
F.S. 180.0
F.S. 215.5
F.S. 248.0
F.S. 263.6
F.S. 300

FIGURE 6-9

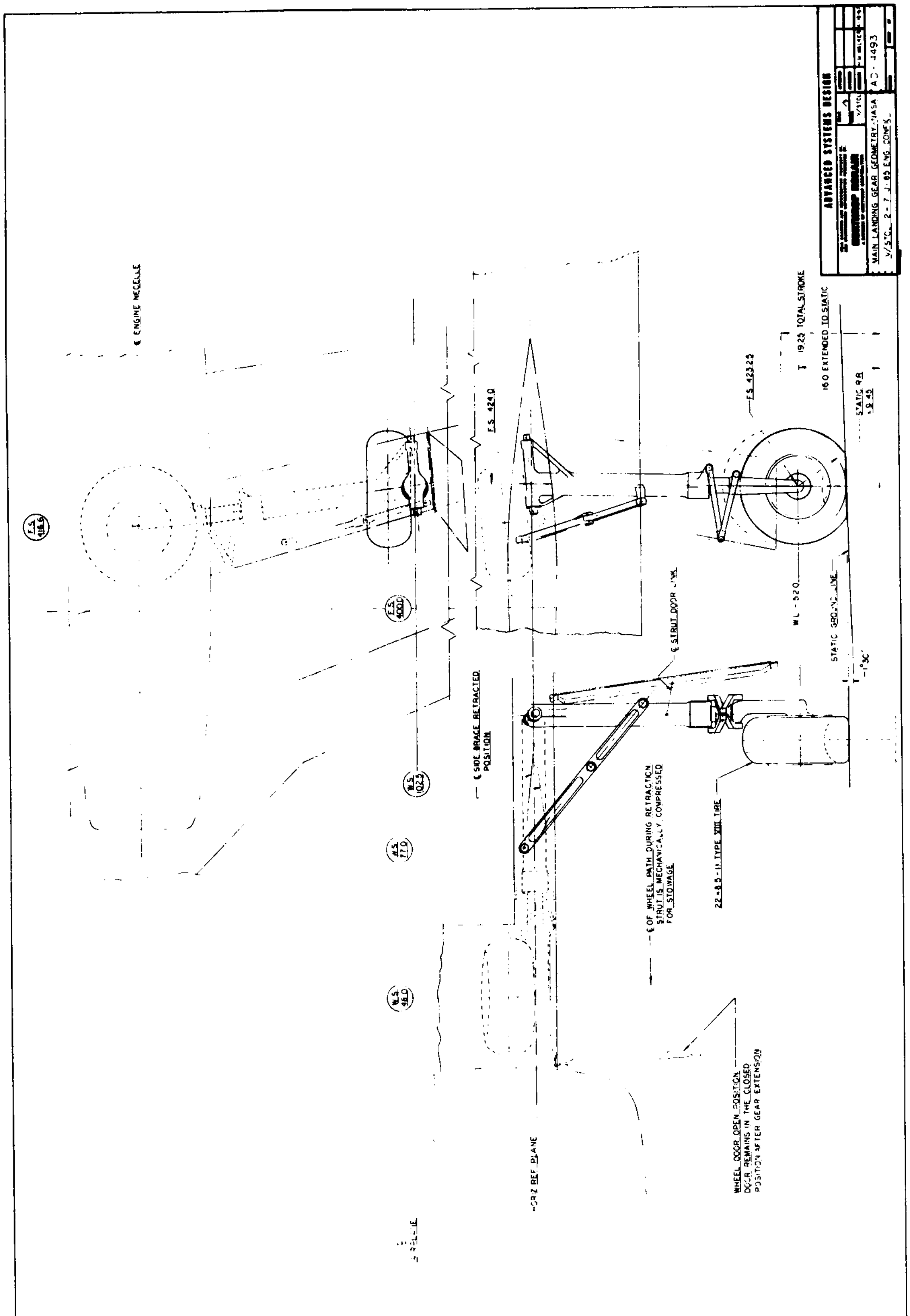

F.S. 416.6
℄ ENGINE NECELLE
W.S. 46.0
W.S. 77.0
W.S. 102.5
F.S. 400.0
℄ SIDE BRACE RETRACTED POSITION
F.S. 424.0
HORIZ REF PLANE
℄ OF WHEEL PATH DURING RETRACTION STRUT IS MECHANICALLY COMPRESSED FOR STOWAGE
℄ STRUT DOOR LINK
22 × 8.5 - 11 TYPE VIII TIRE
WHEEL DOOR OPEN POSITION DOOR REMAINS IN THE CLOSED POSITION AFTER GEAR EXTENSION
F.S. 423.25
WL - 52.0
19.25 TOTAL STROKE
16.0 EXTENDED TO STATIC
STATIC GROUND LINE
STATIC RR 19.45
ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
MAIN LANDING GEAR GEOMETRY-NASA
V/STOL 2-7 J-85 ENG CONFIG
AD-4493

FIGURE 6-10

Note: Gear actuation cylinders and shock strut compression mechanism are not shown on either the main landing gear arrangement Drawing AD 4493 or the nose landing gear geometry, Drawing AD 4494 at this phase of the design; however, adequate space was provided for their design and installation.

6.7 MODIFIED T-39A LANDING GEAR GENERAL DESCRIPTION

The modified T-39A landing gear arrangement as shown on Drawing AD 4458 (Figure 6-11) is of tricycle configuration, utilizing a modified A-4E nose gear strut and wheel assembly and modified A-4E main gear struts with two Northrop T-38 22 x 4.4 tires, wheels and brake assemblies mounted side-by-side on each main strut. All gears are hydraulically actuated and are mechanically compressed during retraction. Gear doors are hydraulically actuated and locks are provided for both the open and closed positions. The landing struts will be equipped for load sensing instrumentation to the cockpit to permit pilot checkout of reaction controls prior to takeoff. A curve of allowable landing sink rate versus aircraft weight is shown on Drawing AD 4458.

6.8 AIR CONDITIONING AND ANTI-ICING EQUIPMENT

An air conditioning system shall be installed with heating capacity sufficient to maintain 65 F minimum in the crew compartment at 0° OAT at pressure altitudes up to 25,000 feet at normal rated engine power. The system shall also maintain 85 F maximum in the crew compartment at an OAT of 110 F at sea level. Compressed bleed air shall be used to energize the system. W/S defrost shall be accomplished by conditioned air vents only. RAM air shall be provided for ventilation when required with the system inoperative. Pressurization shall not be provided. Ground air conditioning with the engines inoperative shall be available with the use of an external compressed air supply. Conditioned air will be exhausted from the cockpit into the avionics bay for component cooling.

The AiResearch Division of the Garrett Corporation manufactures air conditioning systems for Canadian T-37 (CL-41) which uses J-85 engines, and the North American T-39. Either one of these systems appears adequate to meet requirements for the NASA/VTOL research airplane.

6.9 ELECTRICAL LOAD ANALYSIS

The lift/cruise engines are required for hydraulic power and as a pneumatic supply for the air conditioning system as well as electrical power during cruise flight.

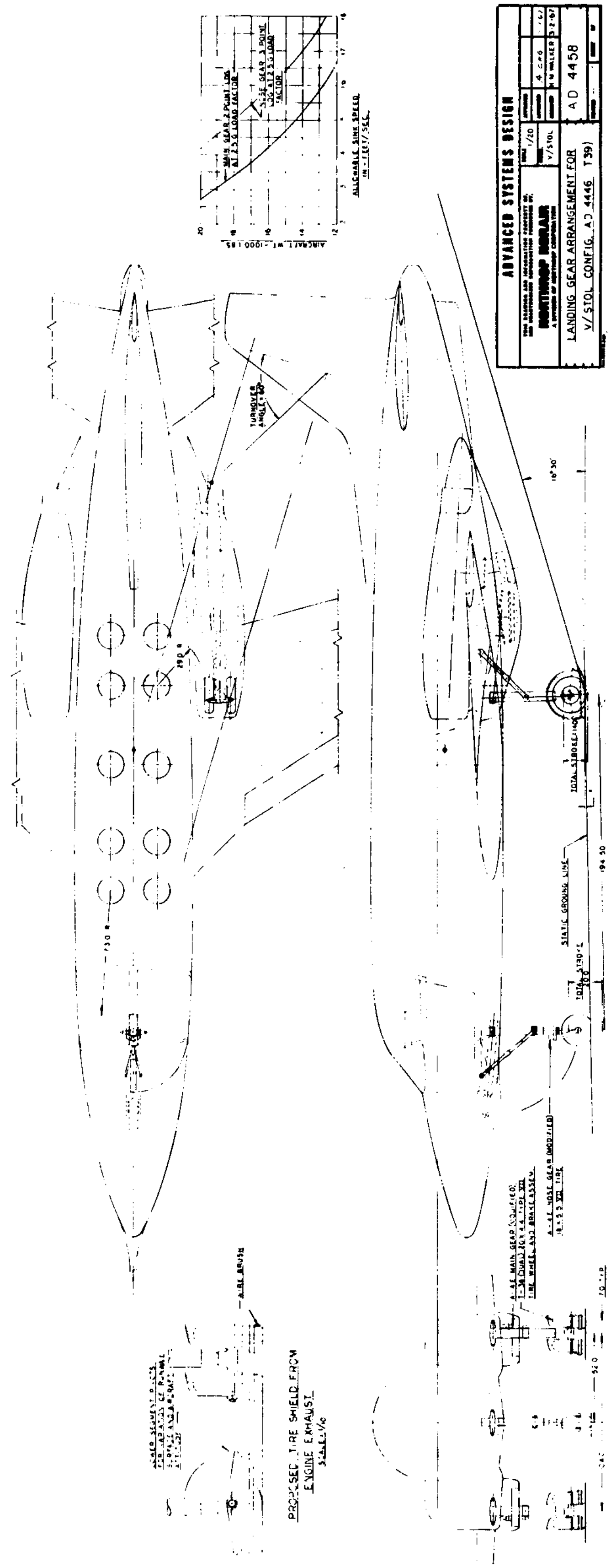

ADVANCED SYSTEMS DESIGN
NORTHROP NORAIR
LANDING GEAR ARRANGEMENT FOR
V/STOL CONFIG. AD 4446 T39)
AD 4458
TURNOVER ANGLE
STATIC GROUND LINE
PROPOSED TIRE SHIELD FROM ENGINE EXHAUST
AIRCRAFT WT - 1000 LBS
ALLOWABLE SINK SPEED

FIGURE 6-11

It is intended that all of the electrical power required be supplied by two fuselage-mounted gear box/generator units, driven by the two lift/cruise engines.

Table 6-2 shows the estimated airplane's electrical power requirements, which gives assurance that the fuselage-mounted gear boxes to be used on the Canadian CF-5 can provide a surplus of electrical power to cover any conceivable growth requirement. These units will be available when the NASA/VSTOL research airplane is built, and will provide constant frequency 400 cycle AC. Auxiliary equipment will be required by the VSS computer to prevent switching transients following a generator failure if it is determined that this is desirable. An electrical system schematic is shown in Figure 6-12.

6.9.1 Emergency Power

A 5-AH battery is required to provide power for the engine igniter system and to energize instrumentation required for engine starts. Emergency cockpit lighting and certain control functions will also remain operational despite a two-engine or two-generator failure.

TABLE 6-2. ESTIMATED ELECTRICAL POWER REQUIREMENTS

	AC WATTS	DC WATTS
AVIONICS		
VORTAC	20	157
IFF		66
ILS		20
VHF COMMUNICATIONS		290
INTERCOM		14
HEADING REF	105	
SWITCHING RELAYS		14
RADIO INTERCONNECT BOX		140
SAS AND VSS	1438	312
ELECTRICAL		
FUEL PUMPS - 9 J85 (4BP)	3300	
FLAP ACTUATORS - T.E. (2)	1600	
- L.E. (1)	1600	
FUEL QUANTITY SYSTEM	40	
FUEL FLOW SYSTEM	20	
INST. AUTOTRANSFORMER	12	
EXHAUST GAS TEMP INDICATOR	72	
CONSOLE LIGHTING AND UTILITY	150	
INSTR. LIGHTING	156	
OXYGEN QUANTITY INDICATOR	72	
CAUTION WARNING SYSTEM	56	
PITOT HEATER	186	
THROTTLE ACTUATORS	50	
MISCELLANEOUS VALVES AND SOLENOIDS	250	
CONTROL RELAYS	50	
INSTRUMENTS	200	
INTERNAL LIGHTING	120	
NASA REQUIREMENTS (ESTIMATE)	1500	
FLIGHT TEST INSTRUMENTS (ESTIMATE)	500	
TOTAL	11497	1013
DC TO AC 1013/.85	1190	
TOTAL AC WATTS	12687	
TOTAL AC VA 13017/.85	14900	
CAPACITY CF-5 GENERATORS	15000 VA EACH	

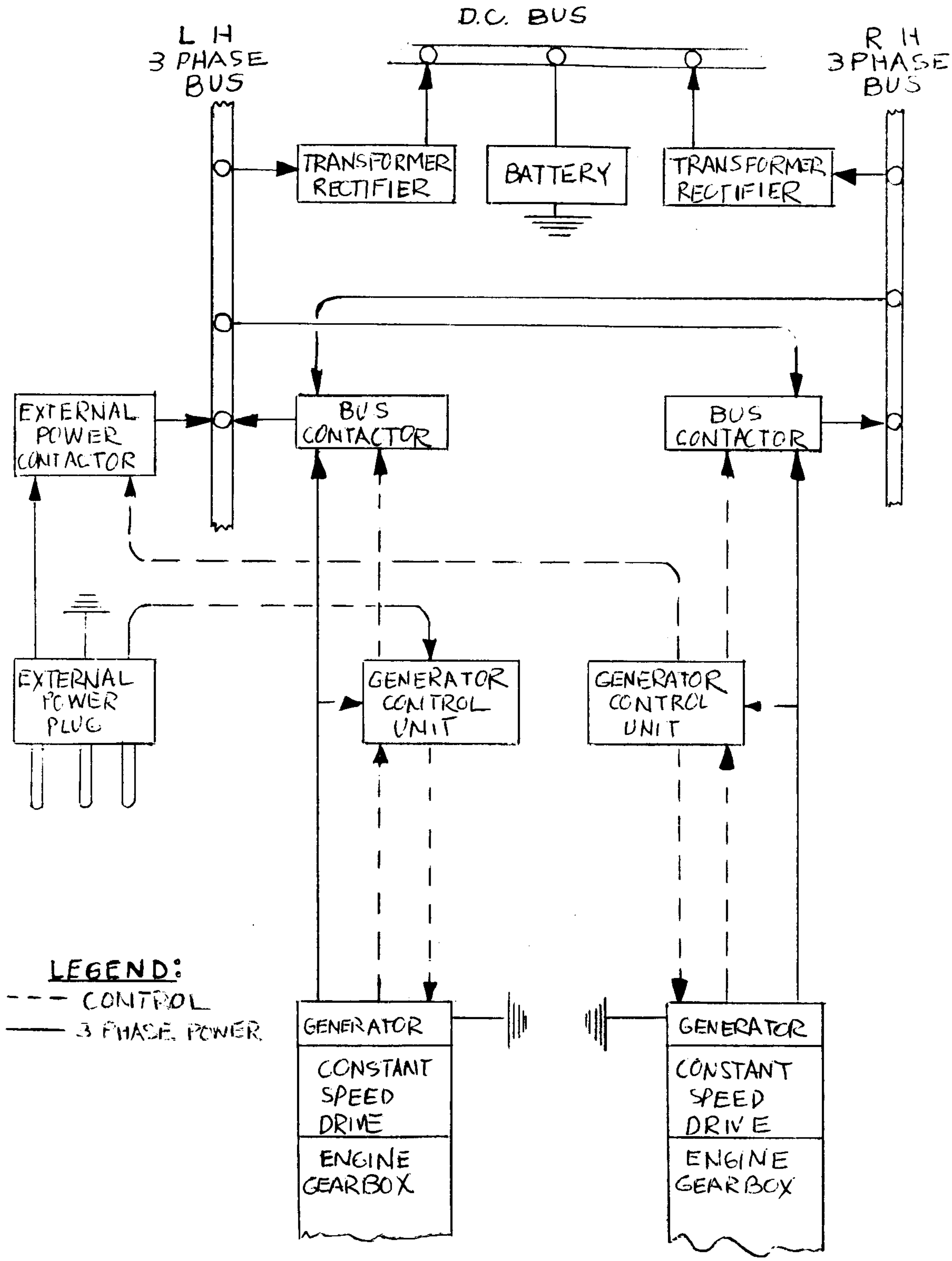


FIGURE 6-12. ELECTRICAL SCHEMATIC

7.0 FLIGHT CONTROLS, SAS, AND VSS SYSTEMS

7.1 GENERAL FLIGHT CONTROLS DESCRIPTION

The primary flight control elements in each cockpit will consist of conventional stick and rudder pedals with provisions for a side stick controller in the evaluation pilot's cockpit.

PITCH CONTROL

The evaluation pilot's control stick will be fitted with a spring, damper and trim actuator for artificial feel. The mechanical advantage of the spring will be adjustable, and the damping ratio will be ground adjustable. Control commands will be transmitted through a pickoff to the VSS system.

A connecting link to the safety pilot will be installed in the aircraft. The schematics and diagrams of the pitch control system show this link as a removable link. However, at very little extra trouble this link can be in-flight connected by means of a switch-controlled engagement. This could occur whenever the VSS is disengaged as an automatic sequence, or could be independently controlled. Manual control plus SAS, manual alone, Fly-By-Wire-Direct or various combinations of these would then be available to the evaluation pilot. Increased safety and versatility should make this attractive.

The safety pilot's control mechanism will be fitted with a trimmable feel system but no damper and no VSS pickoff. Push rods, cranks and control cables will connect the safety pilot's control to the surface and nozzle actuation system and to the parallel VSS servo actuator. This actuator will accept commands through an electric transfer valve. Safety pilot override capability is achieved by overpowering the VSS actuator engage clutch. If no override force is applied by the safety pilot, the actuator moves the output actuators and the safety pilot stick in a manner identical to that customarily associated with a parallel autopilot system. While it is obviously possible to use a force pickup on the safety pilot's stick to attenuate or otherwise modify the VSS command, this feature does not seem necessary for safety.

It should be noted that there is no provision for the evaluation pilot's stick to follow the safety pilot's command, unless the mechanical link is connected.

ROLL CONTROL AND YAW CONTROL

The design philosophy and the type of mechanization for the roll and yaw control systems is similar to the pitch system.

Inertial or gyroscopic coupling may require a scheduled interconnect between the mechanical elements of various axes to avoid overloading the VSS or SAS system, or, to minimize the authority provided the SAS actuators. Pitch trim may require a mechanical connection from the flaps, landing gear or other configuration change; and yaw-roll or other aero coupling may modify the downstream mechanization. Need for these provisions has not been established.

SECONDARY FLIGHT CONTROLS

The leading and trailing edge flaps will be electrically actuated. Flexible torque shafting interconnecting jacks and the actuator will insure symmetry. The leading edge flaps will be scheduled to go to the full down position when the trailing edge flaps are at either the approach or full down position.

7.2 GENERAL SAS DESCRIPTION

The Stability Augmentation System (SAS) is intended to augment the basic VTOL aircraft stability in pitch, in yaw, and in roll. A dual rate feedback system with a fixed compensation and a variable gain will define the SAS complexity. See Figure 7-1. The gain is programmed as a function of $\bar{q}$, dynamic pressure. The rate feedback is sensed by two gyros in each channel (a total of six gyros will be needed for all three channels). To preserve reliability and independence from the Variable Stability System (VSS), the SAS will be designed as a separate unit. However, in order to reduce the VSS cost, some external components, such as rate gyros, will be shared between the two systems. This will not reduce the SAS reliability, or affect its basic independence.

Both gains and compensations of the SAS are designed to satisfy handling qualities requirements during hover (±35 knots), transition (up to 180 knots), and conventional flight (up to M = 0.8, h = 25,000 feet). Recommended AGARD 408A guidelines as modified by NASA will be used for hover and transition flights, MIL-F-8785 for

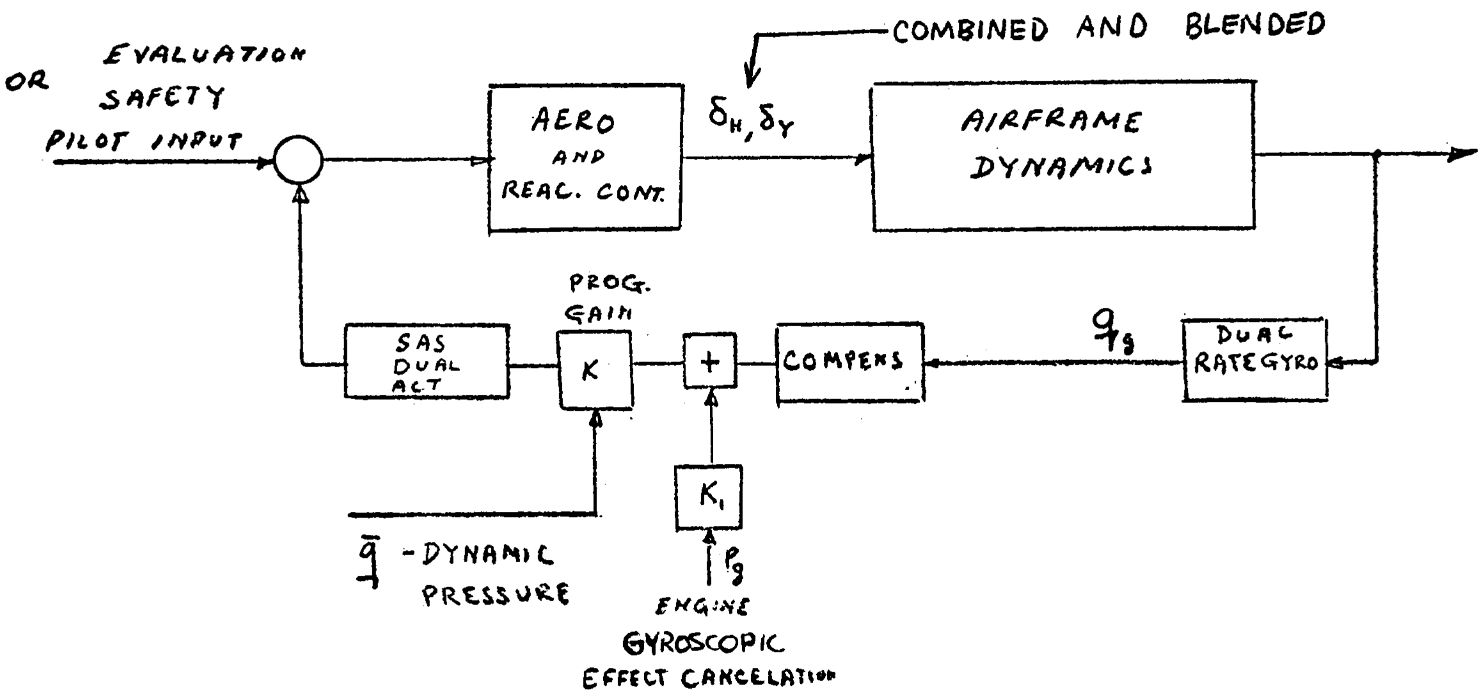


FIGURE 7-1. PITCH SAS CHANNEL

conventional flight. (Norair's interpretation of NASA modified AGARD 408A is shown in Appendix 7-I).

Because of predicted high gyroscopic effect, the pitch and roll SAS channels will use cross-coupling feedback. This cross-coupling feedback will be removed at the end of transition.

7.3 VARIABLE STABILITY SYSTEM

7.3.1 General Description

The Variable Stability System (VSS) is intended for airborne hover and transition simulation of various types of V/STOL aircraft. Basically, the proposed VSS will be a model following type with some forward and feedback compensations. *Six degree of freedom equations of motion, which simulate the dynamics of the selected V/STOL fighter, will be solved by an airborne computer. Five degrees - pitch rate, roll rate, yaw rate, forward velocity, and vertical velocity, however, will be used to drive and force the basic airplane to follow the model.

Figure 7-2 shows the pitch channel in VSS mode. During this period, the evaluation pilot (front cockpit) is in command of the aircraft, while the safety pilot is in standby (rear cockpit). The cockpit controls of the evaluation pilot are "mechanically" disconnected from the aircraft primary controls. The only connection is through electrical wires, which implies that the pilot "flies by wire" only (F.B.W.). The electrical pickoffs (on the cockpit controls) transmit the pilot commands directly to the airborne computer. A number of sensors (such as angle of attack, altimeter, etc.) also supply information to the computer. These sensors supply the computer with fundamental information, mostly about the motions generated by the basic airplane.

A command signal is created as soon as the equations of motion of the simulated fighter are solved in real time by the computer. In the pitch channel, as Figure 7-2 shows, this commanded signal is defined as a pitch rate, q. When the pitch rate is compared with the gyro output, q_g, a definite error signal is generated.

The error signal is amplified and then used through the VSS actuator to drive the primary controls of the basic airplane.

*NOTE: 6-degrees were used for computer sizing. 5-degrees will be mechanized.

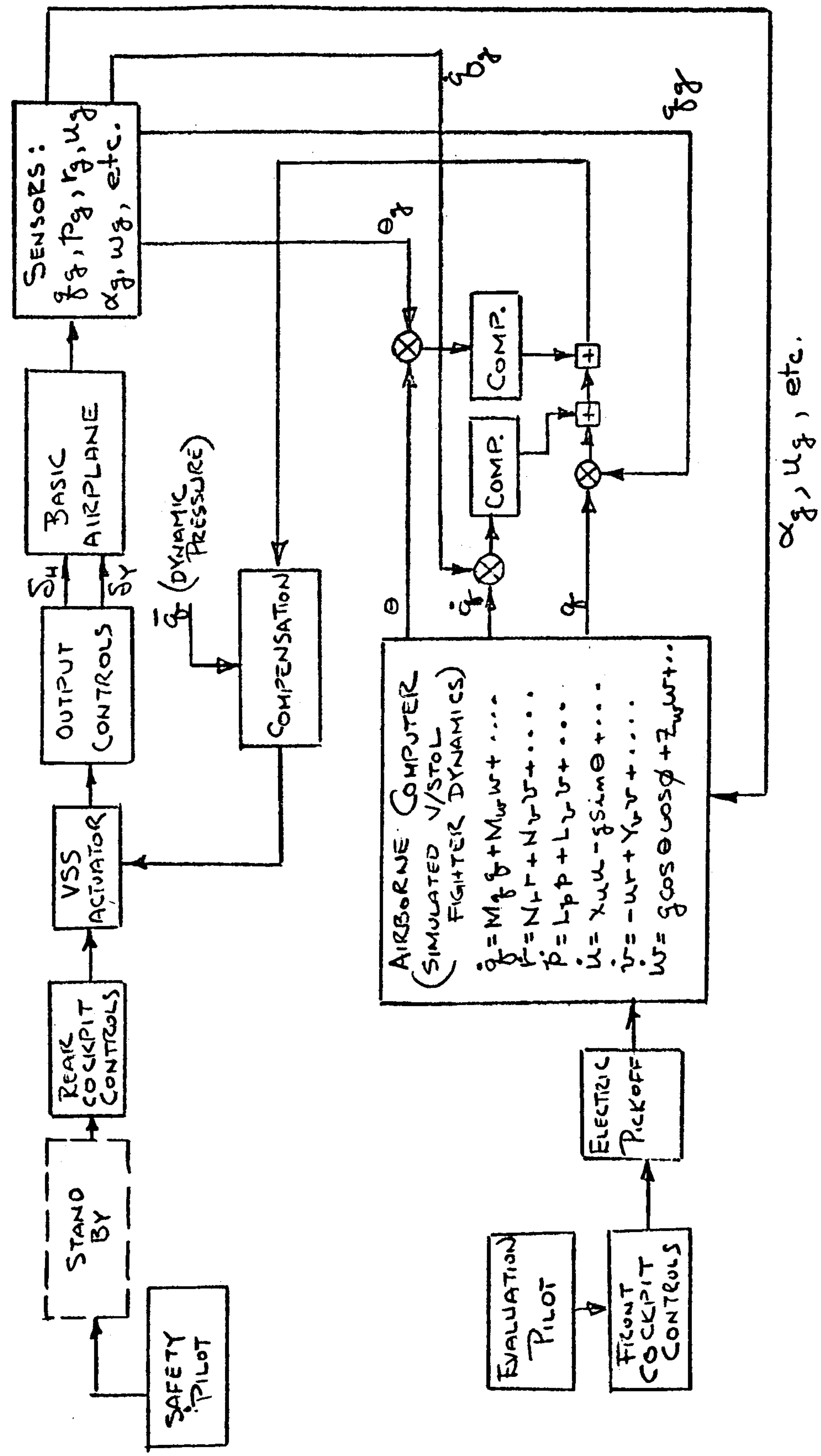


FIGURE 7-2. VSS PITCH CHANNEL

To overcome unavoidable lags in the computer and the control system, the model following technique will require some form of compensation. The basic technique selected to achieve this depends on forcing a match between sensed and computed acceleration and attitude as well as rate. In general, the acceleration match will force the high frequency response and the attitude match the low frequency response. This method will effectively reduce high frequency lags or low frequency attitude drifts which would otherwise exist between the computed pitch motion and the actual pitch motion.

Because of compensation, the closed loop gain can be substantially increased, but not without programmed gain. Figure 7-2 shows that the gain is programmed as a function of $\overline{q}$, dynamic pressure.

The VSS performance can be defined by the delay which can be measured between the simulated and the basic rates in real time. Efforts should be directed to reduce this delay to 0.1 second or less.

7.3.2 Airborne Computer

The heart of the VSS is the digital airborne computer. The computer will be mechanized so that programming of a VSS V/STOL simulation will be greatly simplified. The computer will be composed of a flexible and a fixed portion. See Figure 7-3. The fixed portion of the computer will perform sensor correction and sensor switching; VSS - autopilot modes and associated logics; and signal cancellations. The flexible portion of the computer will be used to simulate a variety of control dynamics; the range of V/STOL jet airplane dynamics (6-deg. of freedom capability); to perform forward and feedback signal simulation and compensation - the "wired feedback", thus generating error command signals to the actuators -- the "aircraft muscles"; to make summation of control inputs and sensor outputs -- the "Patch Boards." The "Patch Board", or Variable Stability System Direct (VSSD) mode will always bypass the Equations of Motion, and signals go directly to the actuators. The Variable Stability System (VSS) mode will include the equations of motion and the computed motions and signals go through the optimized "wired feedback" portion before going to the actuators. All gains and compensation are programmed as functions of dynamic pressure and can be adjusted by operator.

7.3.2.1 EQUATIONS OF MOTIONS. The complexity of an airborne computer depends on the equations needed to be mechanized on the computer. The proposed Norair equations of motions are shown in Figures 7-4 and 7-5. Equations for hover are listed in

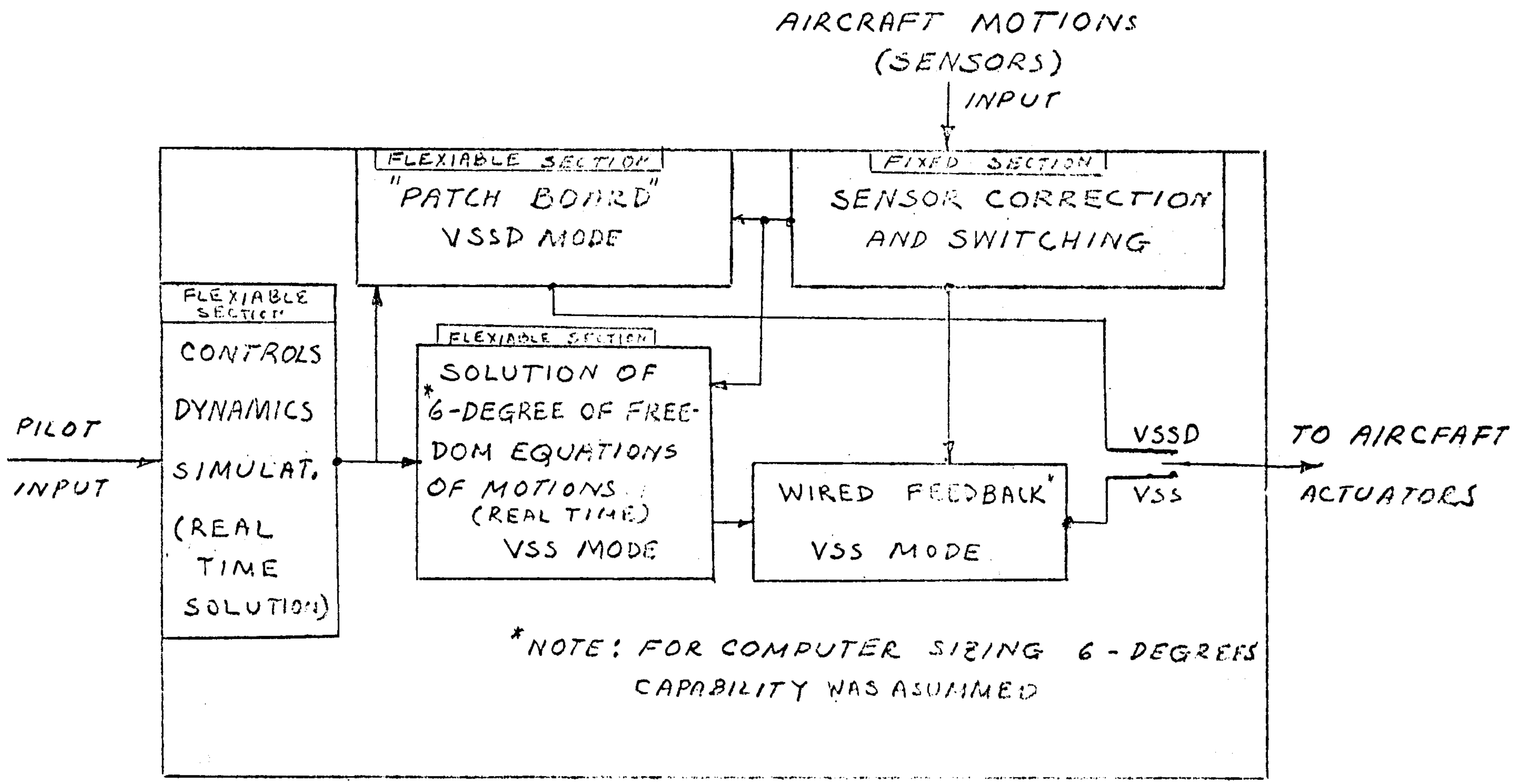


FIGURE 7-3. AIRBORNE COMPUTER

$$(1)\quad \dot{u} = -g\sin\theta + X_u u + X_{\delta_W}\delta_W + X_{\delta_{TC}}\delta_{TC} + X_{\delta_{TL}}\delta_{TL} + X_o\delta_T$$

$$(2)\quad \dot{v} = g\cos\theta\sin\phi + Y_v v + Y_{\delta_Z}\delta_Z$$

$$(3)\quad \dot{w} = g\cos\theta\cos\varphi + \overline{Z_Z H}^{\text{G. EF.}} + Z_w w + Z_{\delta_{TC}}\delta_{TC} + Z_{\delta_W}\delta_W + Z_{\delta_{TL}}\delta_{TL} + Z_{\delta_Y}\delta_Y + Z_{\delta_X}\delta_X + Z_o\delta_T$$

$$(4)\quad \dot{p} = L_p p + L_v v + \overline{L_q q}^{\text{E. CR.}} + \overline{L_{XZ}\dot{r}}^{\text{I.CR}} + \overbrace{L_Z H + L_\varphi \varphi}^{\text{G. EF.}} + \overline{L_\gamma \gamma_2}^{\text{N.IN}} + L_{\delta_X}\delta_X + L_{\delta_Z}\delta_Z$$

$$(5)\quad \dot{q} = M_q q + M_w w + M_u u + \overline{H_Z H}^{\text{G. EF.}} + \overline{M_\gamma \gamma_3}^{\text{N. IN}} + M_{\delta_Y}\delta_Y + M_{\delta_W}\delta_W + M_{\delta_{TL}}\delta_{TL} + M_{\delta_{TC}}\delta_{TC}$$

$$(6)\quad \dot{r} = N_r r + N_v v + \overline{N_\gamma \gamma_4}^{\text{N. ZN.}} + N_{\delta_Z}\delta_Z$$

$$(7)\quad \dot{\varphi} = p$$

$$(8)\quad \dot{\theta} = q$$

$$(9)\quad \dot{\psi} = r$$

UNITS	LEGEND	ASSUMPTION
θ, φ, ψ - in degrees/second	G. EF - Ground Effect	H - Altitude - is measured
q, p, r - in degrees/second	E. CR - Engn. Cross Coup	
$\dot{q}, \dot{p}, \dot{r}$ - in degrees/seconds^2	I. CR - Inert. Cross Coup	
All controls, δ_X, δ_Y, δ_Z, δ_W, δ_{TC}, δ_{TL}, δ_H, δ_A, δ_R, are in degrees	N. IN - Noise or Gust Input	All derivatives contain mass and inertia, for example $X_u = \frac{X_u}{m}$

FIGURE 7-4. HOVER EQUATIONS OF MOTION FOR AIRBORNE COMPUTER (BODY AXIS)

(1) $\dot{u} = -g\sin\theta + X_u(u) + X_w w + X_{\delta_W}\delta_W + X_{\delta_{TC}}\delta_{TC} + X_{\delta_{TL}}\delta_{TL} + X_{o_{\delta_T}}$

(2) $\dot{v} = -(u)\,r/57.3 + g\cos\theta\sin\phi + Y_v v + Y_u(u) - Y_{\delta_R}\delta_R + Y_{\delta_Z}\delta_Z + pw/57.3$

(3) $\dot{w} = (u)\,q/57.3 + g\cos\theta\cos\phi + Z_w w - Z_u(u) + Z_{\delta_H}\delta_H + Z_{\delta_Y}\delta_Y + Z_{\delta_X}\delta_X - Z_{\delta_W}\delta_W - Z_{o_{\delta_T}}$
$\qquad - pv/57.3 - Z_{\delta_{TC}}\delta_{TC} - Z_{\delta_{TL}}\delta_{TL} - \frac{N.IN}{Z_\gamma\gamma_1}$

(4) $\dot{p} = L_p p + L_q q - L_v v + \overset{I.CR}{\overline{L_{XZ}\dot{r}}}\; L_{\delta_R}\delta_R - L_{\delta_A}\delta_A - L_{\delta_X}\delta_X - L_{\delta_Z}\delta_Z - \frac{N.IN}{L_\gamma\gamma_2} + L_{XZ}\,pq$

(5) $\dot{q} = M_q q + M_w w + M_u(u) - M_r r + M_{\delta_H}\delta_H - M_{\delta_Y}\delta_Y + M_{\delta_W}\delta_W - M_{\delta_{TC}}\delta_{TC} + M_{\delta_{TL}}\delta_{TL} - \frac{N.IN}{M_\gamma\gamma_3} - M_{XZ}\,p^2$

(6) $\dot{r} = N_r r + N_v v + N_q q + N_{\delta_R}\delta_R + N_{\delta_A}\delta_A - N_{\delta_Z}\delta_Z - \frac{N.IN}{N_\gamma\gamma_4} - N_{XZ}\,\dot{p}$

(7) $\dot{\phi} = p - \dot{\psi}\,\theta/57.3$

(8) $\dot{\theta} = q - r\phi/57.3$

(9) $\dot{\psi} = r + q\phi/57.3$

UNITS	LEGEND		NOTES
θ, ϕ, ψ - in degrees	I. CR	- Inert. Cross Coup.	All derivatives contain mass or inertia.
q, p, r - in deg/sec	N. IN	- Noise or Gust input	
$\dot{q}, \dot{p}, \dot{r}$ - in deg/sec^2	$\delta_H, \delta_A, \delta_R$	- Aero Contr.	Example:
u, w, v - in ft/sec	$\delta_Y, \delta_X, \delta_Z$	- Reaction Contr.	$X_u = \frac{X'_u}{m}$
$\dot{u}, \dot{w}, \dot{v}$ - in ft/sec^2	δ_W	- Nozzle Contr.	
All controls in degrees	δ_{TL}, δ_{TC}	- Engn. Throt. Con.	

FIGURE 7-5. TRANSITION EQUATIONS OF MOTION FOR AIRBORNE COMPUTER (BODY AXES)

Figure 7-4, and equations for transition in Figure 7-5. All definitions of the variables are listed in Appendix 7-II. The equations contain dimensional derivatives (Section (Section 2.4.4) which already include mass or moment of inertia terms. All rotational variables are expressed in degrees. These equations were simplified for hover and for transitional flights. Where it was possible, the cosine was replaced by 1, sine was expressed as an angle. The cross coupling due to engine effects was included because of significant magnitude of these effects during hover. Since some VTOL aircraft have significant cross coupling in moment of inertia, Ixz, the crosscoupling due to this term was included also. The equations allow for simulation of ground effect, have provisions for simulated gust input and generally contain all important linear and non-linear terms.

Figures 7-6 and 7-7 list the number of function generators which may be utilized during the simulation.

7.3.2.2 COMPUTER MECHANIZATION. The equations of motion will be mechanized by the method which is most practical for the airborne simulation. In the proposed method the dimensional derivatives are specified for fixed velocities at some discrete points of the flight trajectory. The discrete trajectory points may be chosen at $V=0$, $V = 50_{kt}$ $V = 100_{kt}$, $V = 150_{kt}$. The variation of these dimensional derivatives with speed can be utilized to set and mechanize computer function generators. In this method, the accuracy of simulation depends upon the aircraft flight - on how far the aircraft has deviated from the chosen trajectory (See Figure 7-8 for illustration when calculation of X_u is selected).

To illustrate the simplicity of the above approach, consider another method not recommended. Here the derivative is broken into several parts. Even with the assumption that S, m, C_d and C_L are constant this method will require 1 extra function generator, 2 multipliers, and 4 potentiometers.

7.3.2.3 SELECTION OF AIRBORNE COMPUTER. A digital or analog computer could perform the VSS calculation. Both computers have advantages and disadvantages, which are discussed.

The digital computer is better suited for functions, such as multiplication or division, than for real time integration. Since all input/output devices, such as sensors, cockpit controls pickoffs, etc., are analog, a digital computer will require

X_{δ_W}	-	$f_1(1)$
$X_{\delta_{TC}}$	-	$f_2(1)$
$X_{\delta_{TL}}$	-	$f_3(1)$
Y_{δ_Z}	-	$f_4(1)$
Z_Z	-	$f_5(1)$
$Z_{\delta_{TC}}$	-	$f_6(1)$
Z_{δ_W}	-	$f_7(1)$
$Z_{\delta_{TL}}$	-	$f_8(1)$
Z_{δ_Y}	-	$f_9(1)$
Z_{δ_X}	-	$f_{10}(1)$
L_{δ_X}	-	$f_{11}(1)$
L_{δ_Z}	-	$f_{12}(1)$
H_Z	-	$f_{13}(1)$
M_{δ_Y}	-	$f_{14}(1)$
M_{δ_W}	-	$f_{15}(1)$
$M_{\delta_{TL}}$	-	$f_{16}(1)$
M_{δ_C}	--	$f_{17}(1)$
N_{δ_Z}	-	$f_{18}(2)$
L_Z	-	$f_{19}(2)$
L_φ	-	$f_{20}(1)$

All others are assumed constant (constant multipliers)

f (1) - function of one variable

f (2) - function of two variables

FIGURE 7-6. HOVER SIMULATION LIST OF FUNCTIONS (Not Included g cos , etc.) (EQUATIONS OF MOTIONS)

$X_u(u)$	$f_1(u)$	M_{δ_H}	$f_{17}(u)$	Z_γ	$f_{32}(1)$
X_w	$f_2(u)$	N_r	$f_{18}(u)$	L_{δ_X}	$f_{33}(1)$
Y_v	$f_3(u)$	N_γ	$f_{19}(u)$	L_{δ_Z}	$f_{34}(1)$
$Y_u(u)$	$f_4(u)$	N_q	$f_{20}(u)$	L_γ	$f_{35}(1)$
Y_{δ_R}	$f_5(u)$	N_{δ_R}	$f_{21}(u)$	M_r	$f_{36}(1)$
Z_w	$f_6(u)$	N_{δ_A}	$f_{22}(u)$	M_{δ_Y}	$f_{37}(1)$
$Z_u(u)$	$f_7(u)$	X_{δ_W}	$f_{23}(1)$	M_{δ_W}	$f_{38}(2)$
Z_{δ_H}	$f_8(u)$	$X_{\delta_{TC}}$	$f_{24}(1)$	$M_{\delta_{TL}}$	$f_{39}(2)$
L_p	$f_9(u)$	$X_{\delta_{TL}}$	$f_{25}(1)$	$M_{\delta_{TC}}$	$f_{40}(2)$
L_q	$f_{10}(u)$	Y_{δ_Z}	$f_{26}(1)$	M_γ	$f_{41}(1)$
L_σ	$f_{11}(u, \alpha)$	Z_{δ_Y}	$f_{27}(1)$	N_{δ_Z}	$f_{42}(1)$
L_{δ_R}	$f_{12}(u)$	Z_{δ_X}	$f_{28}(1)$	N_γ	$f_{43}(1)$
L_{δ_A}	$f_{13}(u)$	Z_{δ_W}	$f_{29}(1)$		
M_q	$f_{14}(u)$	$Z_{\delta_{TC}}$	$f_{30}(1)$		
M_w	$f_{15}(u)$	$Z_{\delta_{TL}}$	$f_{31}(1)$		
$M_u(u)$	$f_{16}(u)$				

All others are assumed constant (constant multipliers)

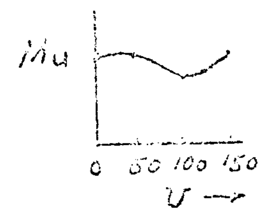
f(1) - means function of one variable

f(2) - means function of two variables

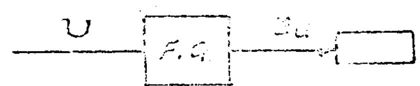
FIGURE 7-7. TRANSITION SIMULATION LIST OF FUNCTIONS (Not Included g cos , etc.) (EQUATIONS OF MOTION ONLY)

1.) Proposed Computer Mechanization

a) Find Functional Variation



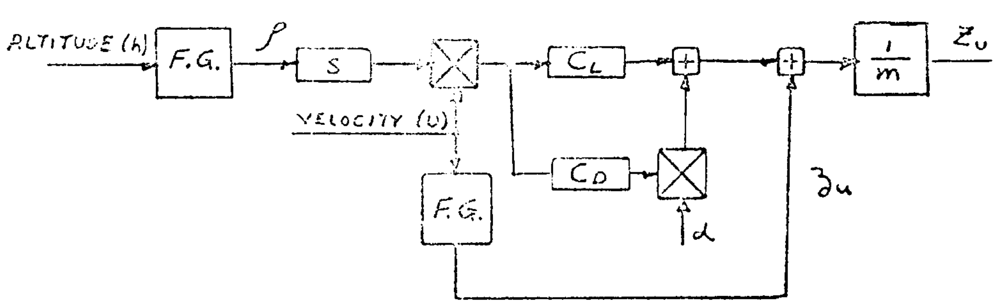
b) Mechanized on Computer



2.) Method Not to be Considered for Computer Mechanization (illustration only)

a) $$Z_u = -\rho \frac{SU}{m}\left(C_L + C_D\alpha\right) + \frac{\mathfrak{Z}_u}{m}$$

b) possible computer mechanization
where S, m, C_D, C_L - constant



where:

-Multipl. -Addition

F.G. -Func. Gen. -Cons. Mult potential

FIGURE 7-8. ILLUSTRATION OF COMPUTER MECHANIZATIONS a large number of digital to analog, and analog to digital conversion units. But there are advantages over analog. These advantages can be listed as follows:

1. To meet future requirements, the computer can be expanded easily.
2. Flight test data recording is easier with a digital computer.
3. The digital computer can be made more accurate than analog.
4. Logical switching and multiplications are well suited for the digital computer.
5. Accuracy of digital computer is almost independent of the power supply.
6. Digital computer has advantages over analog when variation of any dimensional coefficient, or variable, is greater than 1 to 1000.

The analog computer is well suited for the real time integration and summation, but analog multiplication is usually bulky and not very accurate. The overall airborne analog computer accuracy is not better than 5% (providing that ground adjustments are made). The maximum variation of a potentiometer setting is 1 to 1000. However, there are definite advantages over the digital computer. These advantages can be listed as follows:

1. Interface with computer is simple.
2. Analog computer is easily mechanized and any potentiometer setting changes can be made during the airplane flight.
3. Real time integration is well performed.
4. Wiring of analog is simple.
5. Any linear dynamic compensation can be directly programmed.
6. Any variables can be changed easily and independently.

Using the VSS requirement, a survey of the U. S. computer market was made.

About nine computer manufacturers were contacted. All were supplied with basically the same requirements. All were asked to reply with regard to an airborne offshelf computer. The results of this survey are plotted on Figures 7-9 and 7-10. Weight, size and even cost of the analog computer is shown to increase sharply with problem complexity. For the complexity needed for the proposed VSS, the analog weight

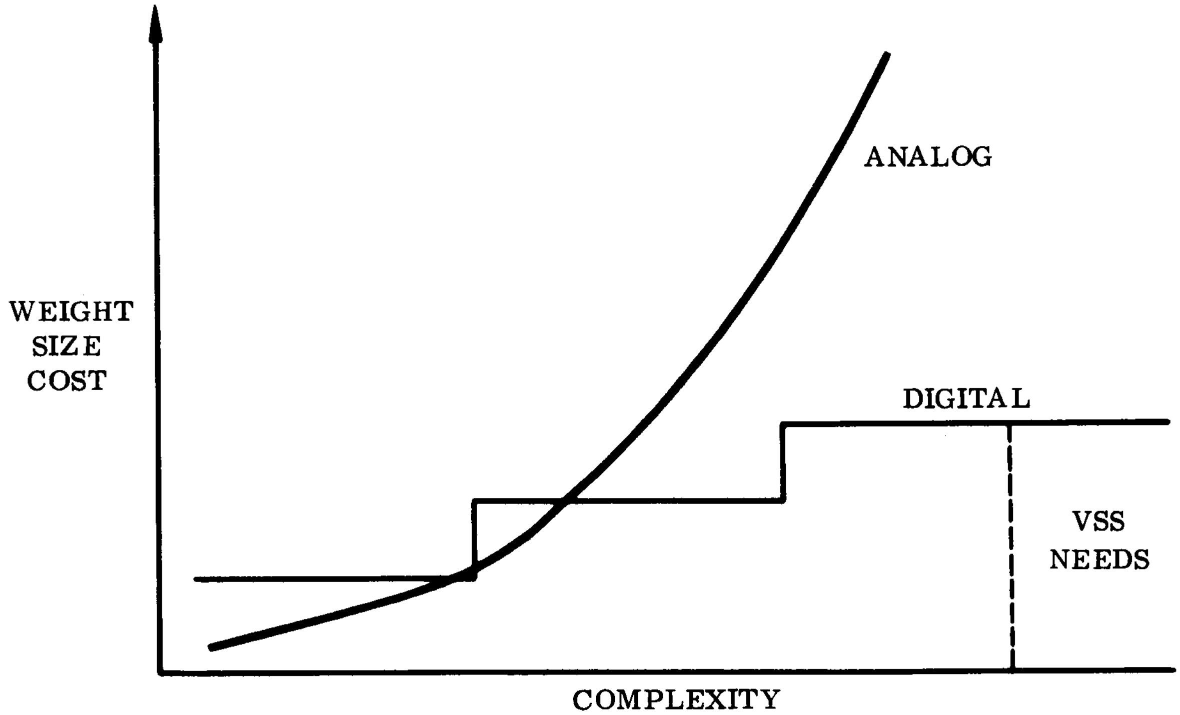


FIGURE 7-9. SURVEY RESULTS (OFFSHELF COMPUTER)

ANALOG (SMALLEST)	DIGITAL (LARGEST)
• SIZE OVER 25 FT^3	2.5 FT^3
• WEIGHT OVER 800 LBS	BELOW 80 LBS

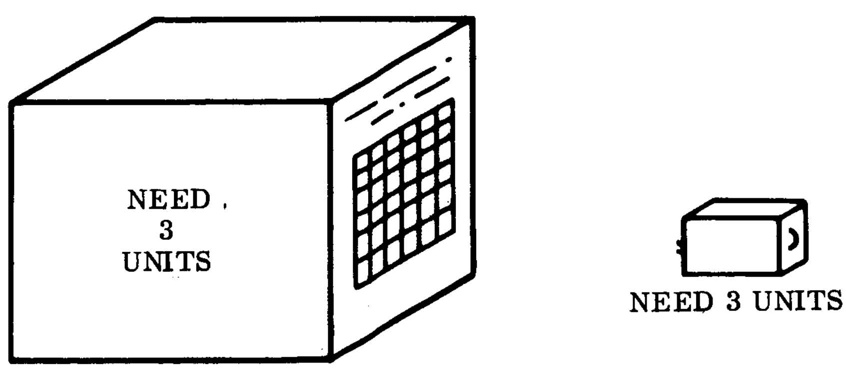


FIGURE 7-10.

goes beyond 800 pounds while the digital computer remains within a manageable 80 pounds. The analog computer volume increases beyond a practical limit also. As a result of this survey, the digital airborne computer has been recommended.

7.3.2.3.1 General Electric vs Teledyne Computer. Two digital computer vendors were considered: General Electric and Teledyne Systems.

Teledyne proposed a computer which was developed under government contract NOw(a)65-0110-f. This computer is specifically designed for a military airborne environment and can be used in VTOL aircraft.

The General Electric computer went through a ten year development period before it reached final form. The computer was specifically designed for flight control application.

The basic summary data for both computers is presented in Figure 7-11. Both computers are incremental type.

The Teledyne computer is a combination of General Purpose (GP) and Digital Differential Analyzer (DDA). The amplitude of the increment can be set by reprogramming, but when programmed, the increment will stay constant during computations. To achieve a high iteration rate (32,000 iterations/second) the machine was designed for parallel type operation.

Unlike Teledyne's computer, the General Electric computer has variable increments of 0, ± 1, ± 2, ± 4, ± 8, ± 16, ± 32, and ± 64. With these larger increments it enjoys a speed advantage of 64:1 over the conventional incremental computer approach. This can be seen in Figure 7-12. The top curve is the response of the General Electric Variable Increment (VI) computer; the middle curve is the response of the Fixed Increment (FI) computer. Both computers here are subjected to step input (magnitude of 100), and both computers are assumed to have the same iteration rate = 1/347 seconds. The assumed iteration rate is actually the complete solution rate used by the General Electric computer with the specified input/output units (Analog/Digital, Digital/Analog). The plotted response of the VI computer has a speed advantage over the FI computer. This can be shown in terms of delay, or equivalent "time constants", T. The estimated T of the VI computer is not more than 0.005 seconds, while T for FP computer is much larger, not less than 0.02 seconds.

Without properly defining A/D and D/A units, Teledyne cannot estimate the final iteration rate (complete solution rate) for the computer. However, if A/D and D/A units design concept is such that it will not cause any appreciable slow down of

VENDOR	TELEDYNE	GENERAL ELECTRIC
Computer Type	Combination General Purpose-Fixed Increment	Special Purpose Variable Increment
Memory Type	a) Lithium Ferrite Core b) Tape Wound Core Rope	a) Metal Oxide Semiconductor b) Wire - Core Read Only (Plug In Module
Clock	750 KC	1 MC and 4 MC
Programming Steps	a) DDA & GP Flow Diagram b) Use IBM 7094 → Punch Paper Tape c) Paper Tape → Ground → Support to Airborne Computer	a) Algorithm Map Flow b) Use IBM 1620 → Specify Core Matrix Table c) Core Wiring of Airborne Computer
Major Computer Units Dimensions	7.5" x 4" x 21" 7.5" x 3" x 21" 7.5" x 3" x 13"	6" x 6" x 17" 6" x 6" x 17" 6" x 6" x 17"
Total Computer Power	440 Watts	170 Watts
Total Computer Weight	74 Lbs	57 Lbs
Total Computer Volume	2.5 Ft^3	1.2 Ft^3

FIGURE 7-11. COMPUTER SUMMARY

the available iteration rate of 32,000/second, the computer response would be of the type the curve shows at the bottom of Figure 7-12.

Another important difference between these computers is programming. Figure 7-11 lists three main steps in the programming of each airborne computer.

The first step in programming the General Electric computer is to set up an algorithm map similar to analog. There is a "cook book" approach for functions such as triple addition, multiply and add, dual integration and addition. These functions (algorithms) can be connected to form an algorithm map. The second step is the conversion of the map into a core matrix wiring table. The IBM 1620 program converts this map into tabular form of easy instructions for each control wire. The third step is actual core wiring. Each control wire is threaded through control point magnetic cores. This exercise is very simple and just requires the following of tabular form instructions. For 1 bit, thread through the magnetic core, and for 0, by-pass the magnetic core. Changes of any given program step can be accomplished by striping the one wire concerned with that program step and rerouting it through the cores to form a different bit pattern. The control core module can be replaced by another pre-wired module and, therefore, can change the "airplane model" in a matter of minutes. In-flight switching can be performed by switching from one module to another.

All the steps in programming the Teledyne computer are different from the GE method. First a flow diagram is formed for the airborne computer instruction. The second step uses an IBM 7094 program to convert these instructions into a paper tape. The third step requires the use of ground support equipment--the test stand. The test stand (in the form of a suit case) will convert the paper tape into airborne computer control instructions by a plug-in to the computer. This approach eliminates manual programming, but always depends on the test stand.

Each computer has both advantages and disadvantages over the other computer. Considering the importance of the computer in the VSS function, it would be wrong to base some kind of selection conclusion on incomplete preliminary computer specifications. With the significant costs involved on each computer, more time devoted to further study of the computer specifications is advised.

It is desirable that each vendor demonstrate the computer programming of an identical sample problem so each computer could be rated on:

1. Ability to perform VSS functions
    a. Computer time delay

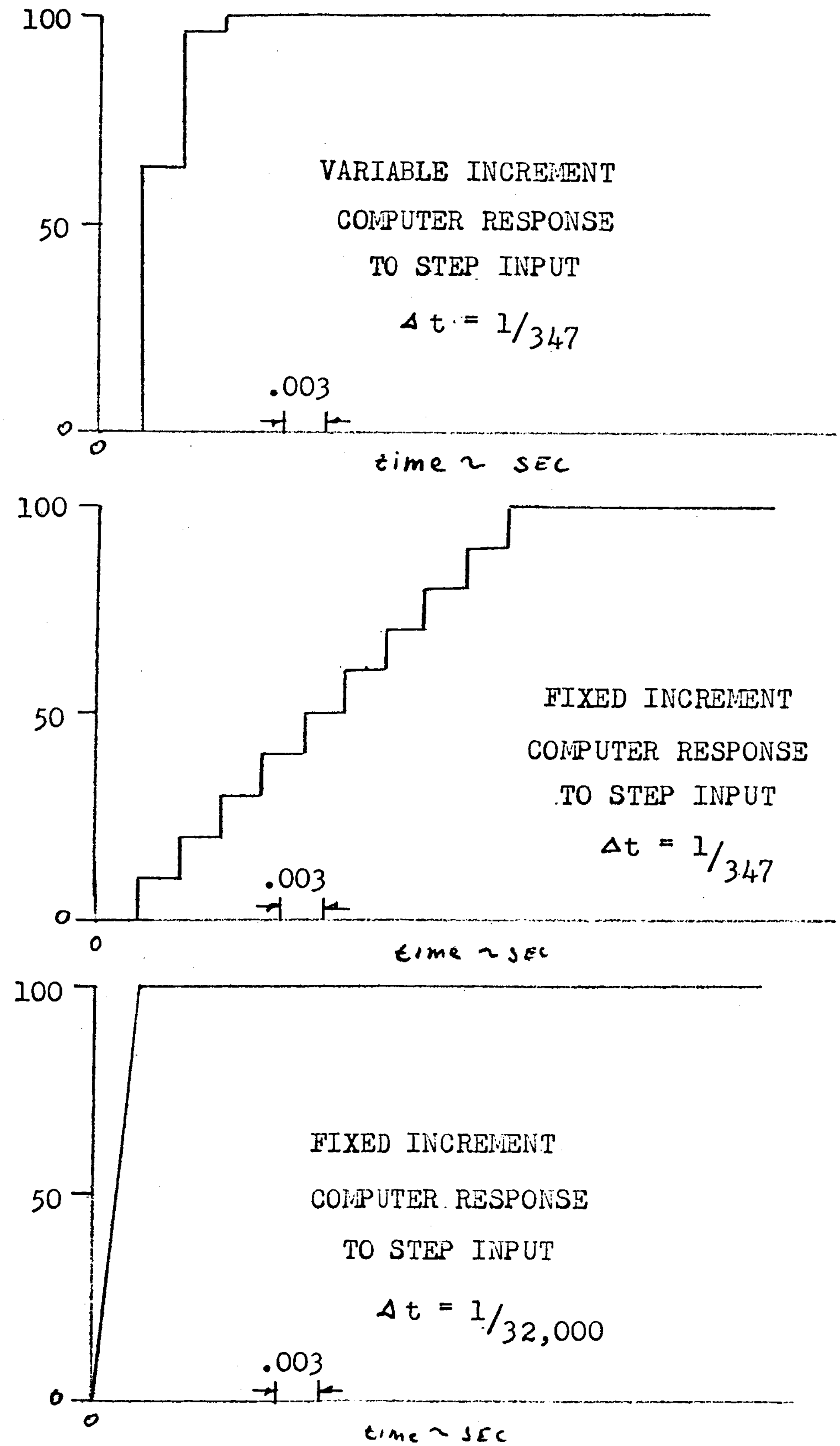


FIGURE 7-12. TIME RESPONSE OF TWO TYPE OF COMPUTERS

b. Input/output scale

c. Solutions to input/output noise problems

d. Open loop integration errors

e. Solutions to "overflow" in DDA

f. Slew-rate limitations

2. Programming

a. More detailed description of steps

b. Approximate time for programming

c. Programmers background (proficient DDA expert)

d. Method of function generator presentation and use

e. Automatic vs manual

f. Computer check and trouble shooting

3. Reliability and Cost

a. More detailed cost definition

b. Previous use of computer in airborne vehicle

c. Cost of replaced modules

d. Estimated reliability

4. Dimensions, weight, power

a. Better power definition (limitations)

b. Weight of total computer

5. Ground support

a. Cost

b. Size, weight, type of power

c. Type of field support (technical, program)

6. Computer Flexibility

a. Computer expansion

b. Computer control

c. Computer check out

7.4 SAS AND VSS CONTROL SYSTEM

7.4.1 Modes of Operation

There are essentially five modes of operation of the proposed VTOL control system:

Direct

Autopilot

Fly By Wire Direct (FBWD)

Variable Stability System (VSS)

Variable Stability System Direct (VSSD)

Direct mode applies to the safety pilot only. In this mode, the pilot flies the airplane with a conventional mechanical connection between cockpit controls and a power actuator (See Figure 13).

Autopilot is an attitude hold and stabilization mode of automatic control of the basic airplane.

Fly by wire direct (FBWD) is an unconventional method of controlling the power actuators. It is a special case of FBW (Fly by Wire). A mechanical connection between a cockpit control and a power actuator is replaced by a "wire" carrying an electrical signal which is proportional to a pilot command. (See Figure 7-13.) In this mode the airborne computer is bypassed.

VSS is also a special case of FBW, but unlike FBWD, the electrical signals are shaped by the "model"--the airborne computer (see Figure 7-13).

VSSD is the "Patch Board" of the airborne computer. (See Figure 7-3.) The pilot commands here are "patched up" with the sensor signals and always bypass the Equations of Motion, and go directly to the actuators. Extreme care in using this mode is recommended due to a constant dependence of the mode on a programmer's skill and knowledge of the basic aircraft dynamics to use proper feedback gains and signs.

Figure 7-14 illustrates the modes of operations and major assumptions of this requirement. As a ground rule, the evaluation pilot will always fly the airplane. There are exceptions to this rule. The safety pilot will fly the airplane during new pilot checkout and during airplane pre-delivery flight test. During emergency, the safety pilot can overpower VSS or autopilot operation.

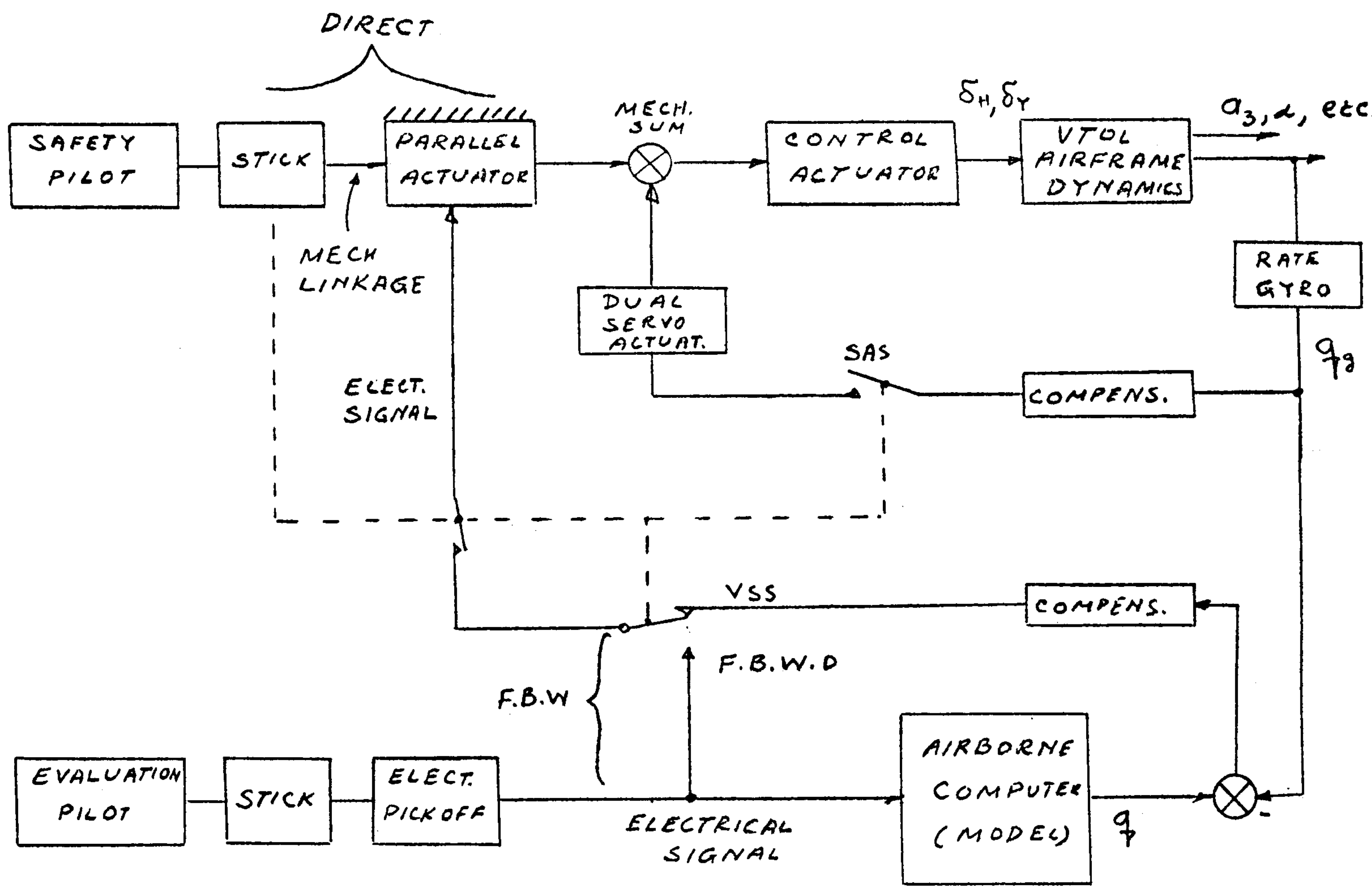


FIGURE 7-13. MODES OF OPERATION (PITCH CHANNEL-SIMPLIFIED)

I. CONDITIONS ASSUMED IN V/STOL STUDY

Type of Flying	Pilot	Mode of Operation	SAS	Note
Normal	Evaluation	F.B.W.D.	On	
	Safety	Stand by		
Normal	Evaluation	VSS or VSSD	Off	Any Stability F.B. are Mechanized in VSS
	Safety	Stand by		
Normal	Evaluation	Stand by	On	Flight Test or Training
	Safety	Direct		
Emerg.	Evaluation	VSS or VSSD	On	VSS Failure
	Safety	Direct - VSS or VSSD is overpowered		
Emerg.	Evaluation	F.B.W.D.	Off	SAS Failure
	Safety	Direct - F.B.W.D. overpowered		

II. CONDITIONS NOT TO BE CONSIDERED IN V/STOL STUDY

Type of Flying	Pilot	Mode of Operation	SAS	Note
Normal	Evaluation	F.B.W.D.	Off	
	Safety	Stand By		
Normal	Evaluation	VSS	On	
	Safety	Stand By		

F. B. - Feed Back

FIGURE 7-14. MODES OF OPERATIONS - ASSUMPTIONS

7.4.2 Control System

7.4.2.1 ROTATIONAL CONTROL (PITCH, ROLL, YAW). The block diagram which is typical for pitch, roll, or yaw channels is shown in Figure 7-15. This figure indicates how the autopilot is functionally related to the airborne computer in the VSS mode. A mechanical implementation is shown on Figure 7-16.

Safety requirements and multiple input capability are the most important factors which dominated the control system definition. Precautions have been taken to make each channel responsive (pitch, roll, and yaw).

The proposed control system has almost identical frequency response for each channel (from cockpit controls to *external control, or from VSS computer output to *external control). This frequency response can be approximated by third order lag, with time constant, $\tau = 0.032$ seconds, and with natural frequency, fn = 5 cps, and damping ratio, $\zeta = 0.2$.

During the transition phase, the airplane moment is controlled by a combination of an aerodynamic control surface input, and reaction control jets. These two kinds of external airplane controls can be controlled either from the front cockpit controls (evaluation pilot) or from aft cockpit controls (safety pilot). During FBWD or Direct mode, the pilot inputs are combined with the SAS inputs.

With these multiple input capabilities several combinations of signals may be assumed; however, only systems which reflect requirements illustrated by the conceptual diagram of Figure 7-17 are to be considered.

The concept is a simple, easily mechanized control system. The VSS electrical signal will energize the parallel actuator, which then will move simultaneously the safety pilot stick, and both external controls. As Figure 7-17 shows, the stabilizer is geared to the reaction control position. The gearing can be fixed, or programmed to respond to a slowly changing variable (trim).

7.4.2.2 TRANSLATIONAL CONTROL (THROTTLE AND VECTOR). The Translational Control System consists of two separate channels: the throttle control and the vector control. Each pilot is supplied with identical cockpit controls, two throttle controls and only one vector control. One throttle control is for the two lift/cruise engines, and another control for all lift engines. Safety and evaluation pilot

*External control is reaction nozzle position, rudder position, etc.

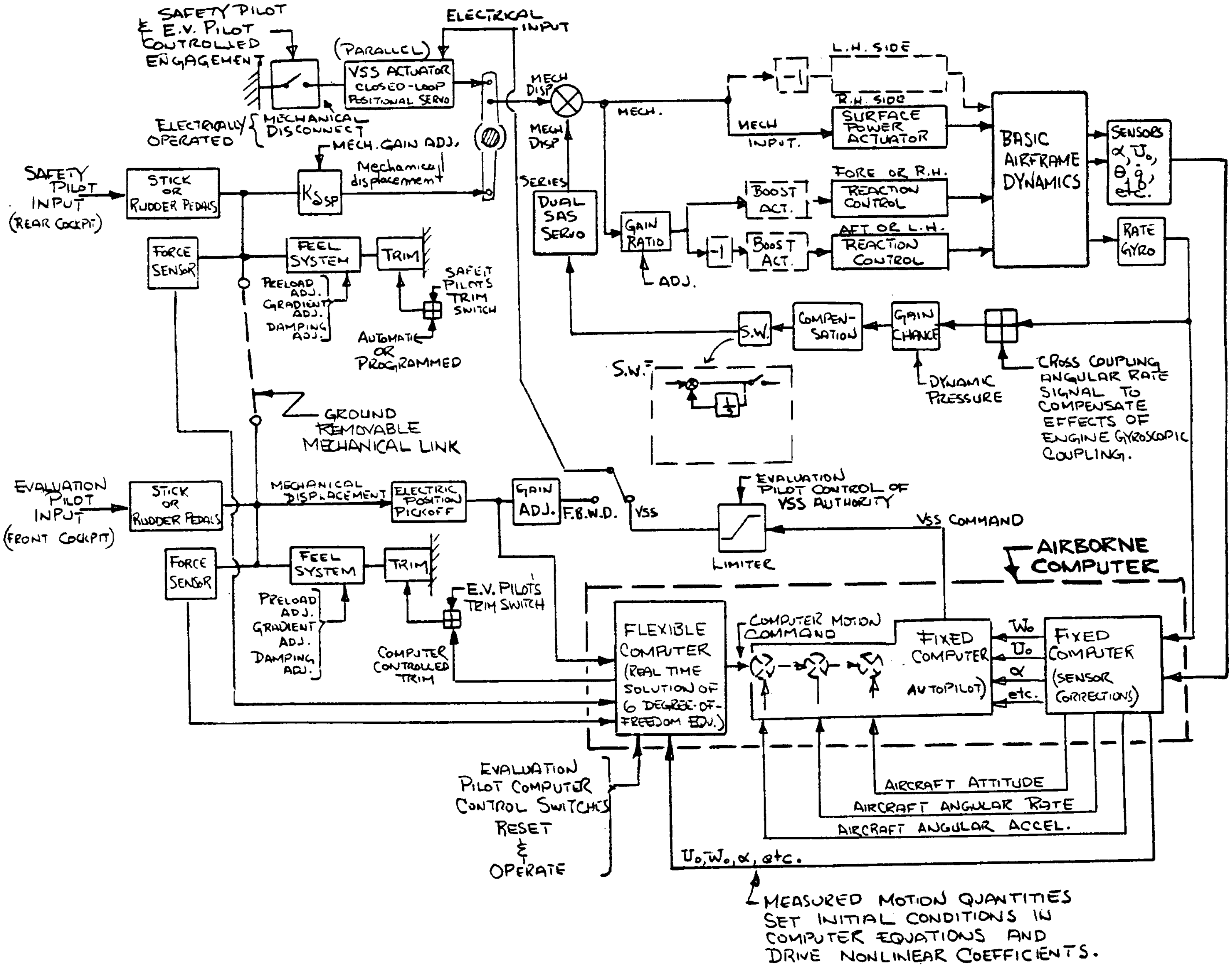


FIGURE 7-15. PITCH, ROLL, OR YAW CONCEPTUAL DIAGRAM

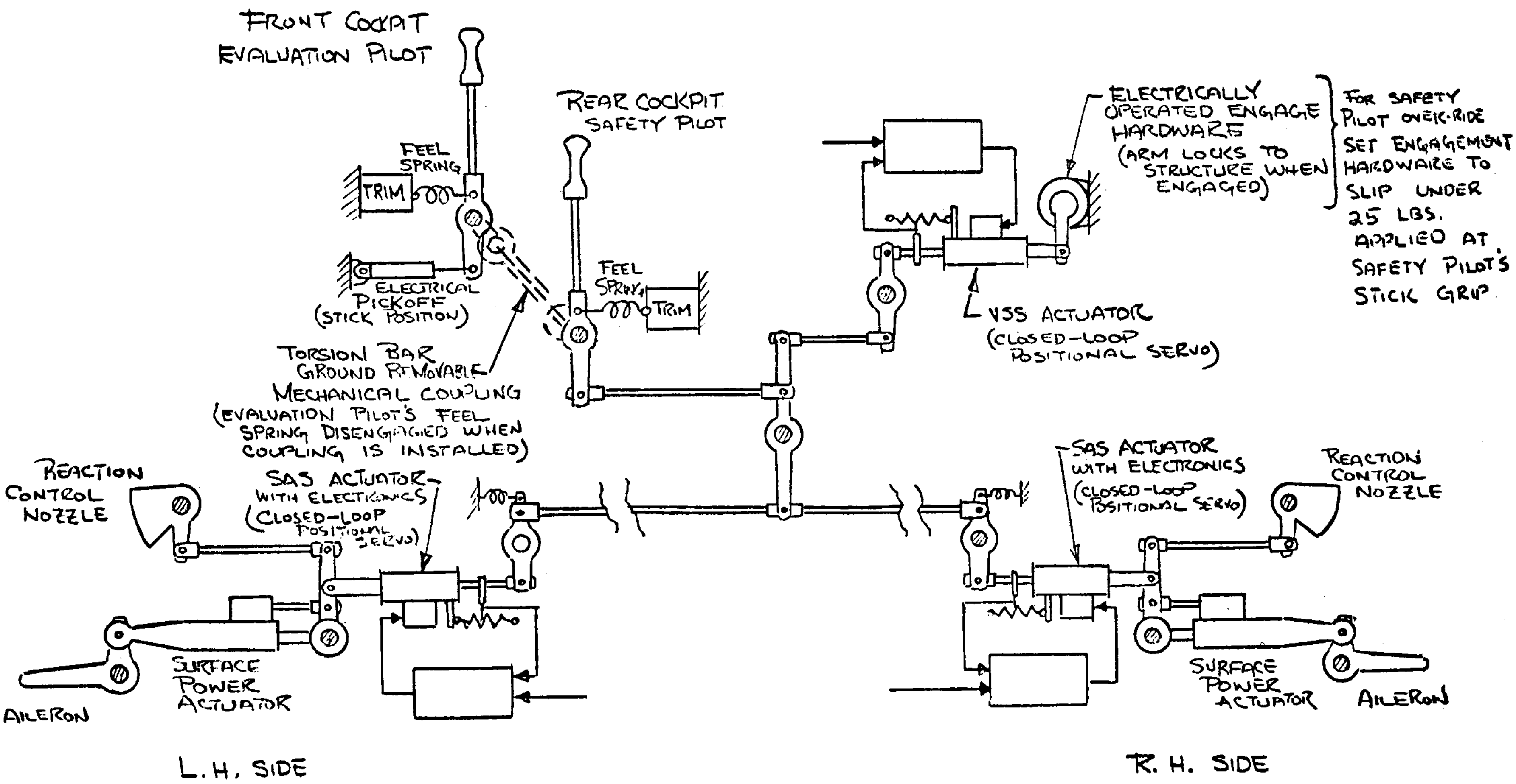


FIGURE 7-16. SCHEMATIC DIAGRAM - ROLL CHANNEL CONTROL SYSTEM CONCEPT

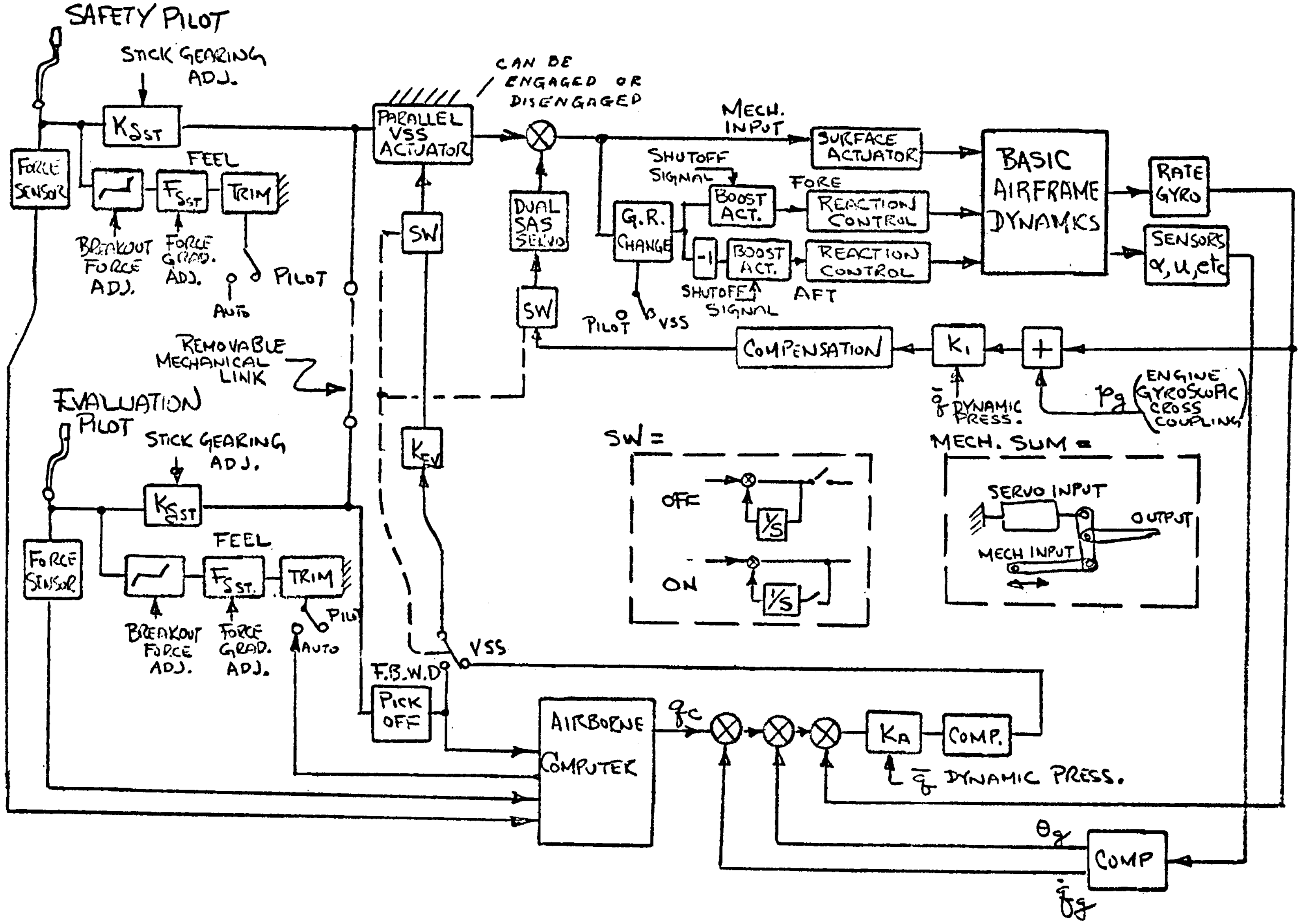


FIGURE 7-17. PITCH CHANNEL CONCEPT

cockpit controls are mechanically disconnected from each other under normal flight conditions. The evaluation pilot electrical signals can be fed into the computer for VSS mode, or bypassed for FBWD mode (see Figure 7-18).

In the throttle channels, the pickoff-signals or computer output drive the parallel actuators so the safety pilot's controls always move. The safety pilot is in standby during normal flight; however, he can always overpower the parallel VSS actuator at any time.

The vector control channel is different from the throttle. The pickoff-signals or computer output do not move the evaluation pilot control. Only a cockpit display will indicate nozzle-vector position. However, the safety pilot can always override the evaluation pilot. A commanded vector input will also energize the latching relay, and immediately open FBWD, VSSD or VSS outputs.

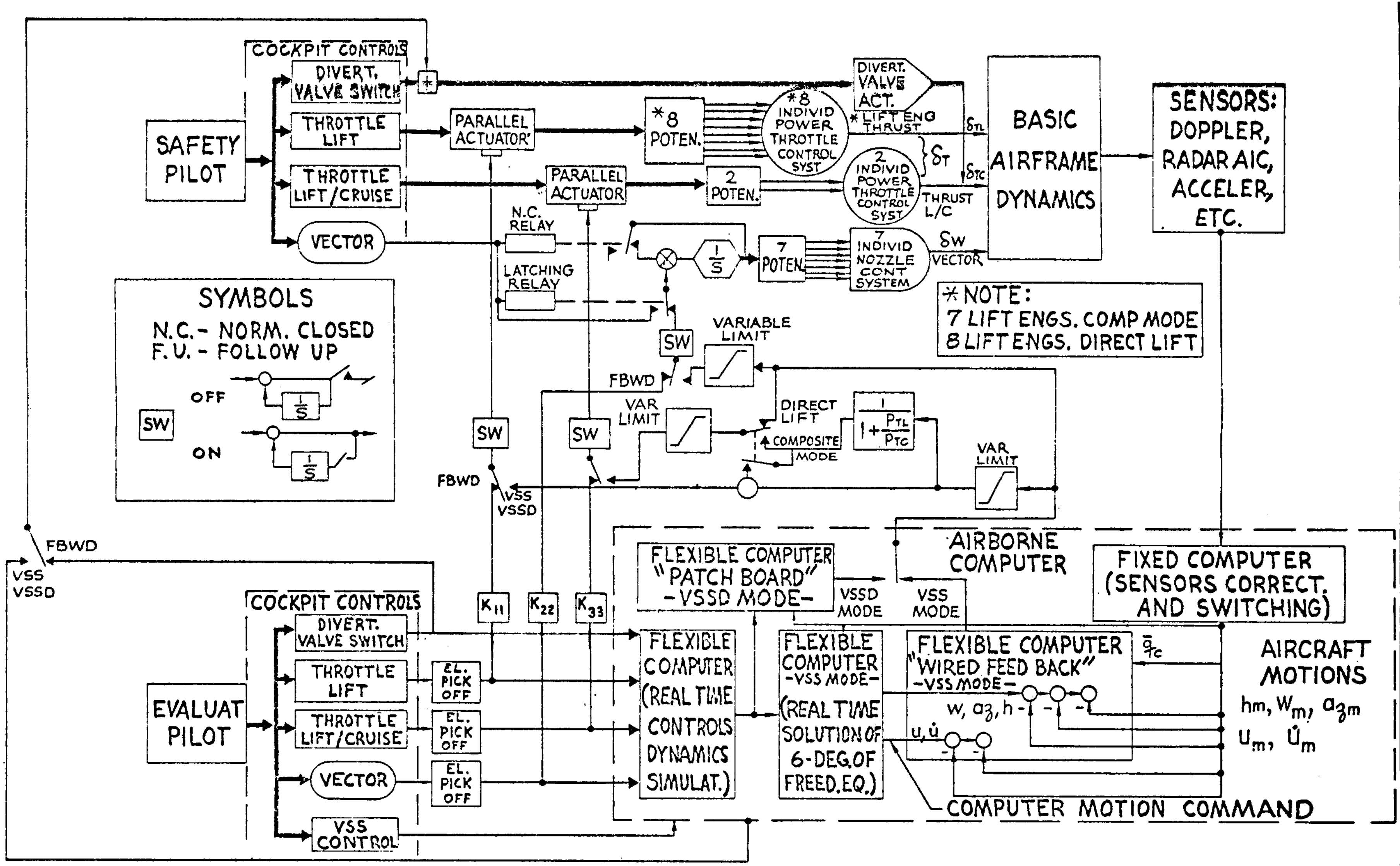
The lift engines will be equipped with movable nozzles and nozzle servo actuators supplied by the engine manufacturer.

The principle of VSS operation is the same as for the pitch channel. Instead of pitch rate, two translational velocities of the basic airplane will be used to match with the simulated airplane velocities, u_c, w_c. The evaluation pilot's input first will be shaped by a "model", the airborne computer, and then compared to the sensed velocites, u_m, w_m. The error signals, between the model and the sensed velocities, then will be used to command an incremental change in nozzle position and in throttle position. This command, if idealized, is that which will produce a time program of x and z forces as necessary to match actual aircraft velocities, u and w, with those of the simulated airplane, u_c and w_c.

The VSS translational control requires sufficient flexibility for operation in either of two modes: direct lift mode or composite mode. The present control mechanization is capable of accomplishing this by a simple switch position.

In the direct lift mode, the lift engines provide all the necessary engine lift forces through hover and transition; in this mode, the two lift/cruise engines are used only for forward thrust.

In the composite mode, the lift engines plus the two lift/cruise engines with downward vectored thrust, will provide all necessary engine lift during hover, and during transition.



10861 YZ

FIGURE 7-18. VSS BLOCK DIAGRAM (THROTTLE AND VECTOR)

The thrust from all lift engines can be vectored (by positioning all nozzles) 15 degrees forward and 28 degrees aft. The expected maximum rate of thrust vectoring is 90 degrees/second.

The safety or the evaluation pilot can fly the basic airplane independently, take off and land, however, the safety pilot can always overpower the evaluation pilot. When the evaluation pilot is flying, except for vector and diverter valves, all the safety pilot cockpit controls are moving. The vector and diverter valve position, however, are always displayed to the safety pilot.

The vector and the throttle channels in each cockpit have one vector control, one diverter valve control, and two throttle controls--one for all lift and one for all lift/cruise engines.

The proposed control system frequency response is as follows:

1. Vector channel is second order lag with natural frequency, fn = 8 cps and damping ratio, ζ, between 0.5 and 1.
2. Throttle Channel
   a. G.E. engine is expected to act a first order lag with time constant, $\tau = 0.2$ second.
   b. The throttle control is third order lag with time constant = 0.05 second and natural frequency fn = 3.5 cps. Damping ratio, ζ, between 0.3 and 0.8.

7.4.3 Switching

Switching from VSS to SAS is made safely, even when VSS is overpowered. This is because all VSS commands synchronize the safety pilot stick motion with both external controls.

Switching from SAS into VSS or from VSS into SAS can be accomplished with very small transient by electrically engaging or disengaging the VSS actuator only after the actuator is trimmed with all input commands active.

7.4.3.1 AUTOPILOT SWITCHING. The requirement for inflight switching (normal flight) is outlined in Figure 7-19.

INITIAL MODE OF CONTROL	CONDITIONS OF INITIAL MODE	DESCRIPTION OF SWITCHING OPERATION (From Initial to Final Mode)	FINAL MODE OF CONTROL	CONDITIONS OF FINAL MODE
SAFETY PILOT FLYING (SAS ON)	1. SAS on 2. E.V. Controls Off 3. Autopilot Off 4. VSS Act. Disengaged	1. Safety Pilot Holds Airplane in Straight and Level Unaccelerated Flight 2. E.V. Pilot Switches E.V. Controls on F.B.W.D. to VSS Act. 3. E.V. Pilot Trims E.V. Controls 4. E.V. Pilot Engages VSS Actuators	EVALUATION PILOT F.B.W.D. (SAS ON)	1. E.V. Controls on F.B.W.D. 2. SAS On 3. VSS Act. Engaged 4. Autopilot Off
SAFETY PILOT FLYING (SAS ON)	(See Above)	1. Safety Pilot Holds Airplane in Straight and Level Unaccelerated Flight 2. E.V. Pilot Closes Autopilot Loops to VSS Actuators 3. E.V. Pilot Engages VSS Actuators;Engagement of VSS Actuators with Autopilot Loop Closed Also Turns SAS Off.	ATTITUDE HOLD (AUTOPILOT)	1. E.V. Controls Off 2. SAS Off 3. VSS Act. Engaged 4. Autopilot On
ATTITUDE HOLD (AUTOPILOT)	1. E.V. Controls Off 2. SAS Off 3. VSS Act. Engaged 4. Autopilot On	1. E.V. Pilot Sets All Initial Conditions into Computer using Measured U_o, α, W_o, etc 2. E.V. Pilot Trims E.V. Controls and Adjusts Voltage Output on E.V. Controls to Zero 3. E.V. Pilot Switches E.V. Controls to VSS Computer and Switches Computer to Operate 4. E.V. Pilot Fades on Computer Output to VSS Actuators	EVALUATION PILOT VSS (SAS OFF)	1. E.V. Controls to VSS Computer On 2. SAS Off 3. Autopilot On 4. VSS Autopilot Command On
EVALUATION PILOT VSS (SAS OFF)	1. E.V. Controls to VSS Computer On 2. SAS Off 3. Autopilot On 4. VSS Autopilot Command On	1. E.V. Pilot Holds Airplane in Straight and Level Unaccelerated Flight 2. E.V. Pilot Fades Off Computer Output From VSS Actuators 3. E.V. Pilot Disengages VSS Actuators; Disengagement of VSS Act. With Autopilot Loop Closed also Turns SAS On	SAFETY PILOT FLYING (SAS ON)	1. SAS On 2. E.V. Controls Off 3. Autopilot Off 4. VSS Act. Disengaged
EVALUATION PILOT F.B.W.D. (SAS ON)	1. E.V. Controls On F.B.W.D. 2. SAS On 3. VSS Act. Engaged 4. Autopilot Off	1. E.V. Pilot Holds Airplane in Straight and Level Unaccelerated Flight 2. Safety Pilot Trims Safety Pilot's Controls For Zero Force on Feel System 3. E.V. Pilot Disengages VSS Actuators	SAFETY PILOT FLYING (SAS ON)	(See Above)
ATTITUDE HOLD (AUTOPILOT)	1. E.V. Controls Off 2. SAS Off 3. VSS Act. Engaged 4. Autopilot On	1. Safety Pilot Trims Out Safety Pilot's Controls for Zero Force on Feel System 2. E.V. Pilot Disengages VSS Actuators; Disengagement of VSS Act. with Autopilot Loop Closed Also Turns SAS On	SAFETY PILOT FLYING (SAS ON)	(See Above)
EVALUATION PILOT VSS (SAS OFF)	1. E.V. Controls To VSS Computer On 2. SAS Off 3. Autopilot On 4. VSS Autopilot Command On	1. E.V. Pilot Holds Airplane in Straight and Level Unaccelerated Flight 2. E.V. Pilot Fades Off Computer Output From VSS Actuators 3. E.V. Pilot Trims E.V. Controls for Zero Output Voltage and Desired Control Position 4. E.V. Pilot Switches E.V. Controls on F.B.W.D. to VSS Actuators 5. E.V. Pilot Fades Out Autopilot Loop to VSS Actuators: This Action Turns SAS On	EVALUATION PILOT F.B.W.D. (SAS ON)	1. E.V. Controls On F.B.W.D. 2. SAS On 3. VSS Act. Engaged 4. Autopilot Off

FIGURE 7-19. NORMAL INFLIGHT SWITCHING REQUIREMENT

7.4.4 VSS Cockpit Controller

The VSS cockpit controller is intended for VSS function control from the front cockpit by the evaluation pilot. The following functions are available.

1. Mode Selector. Two position switch, VSS and VSSD.
2. Switching on and off of all VSS actuators. This switch will actuate all clutches in parallel actuators.
3. Null indicator - optional for evaluation pilot - used during switching from the safety pilot controls to the evaluation pilot controls. This indicator will indicate the pitch stick relative positions.
4. FBWD switch. This switch in ON position and VSS actuator clutch switch to ON position will put aircraft into FBWD mode.
5. Attitude hold. This switch in ON position and VSS actuator clutch switch to ON position will put aircraft into attitude hold mode.
6. The computer operation switches are RESET and COMPUTE.
7. Initial condition pots will set the initial conditions into compute.
8. Flight condition select will change the flight condition in the computer.
9. VSS authority set controls will change signal authority from 30 percent to 100 percent. There are six independent controls.
10. VSS mode engagement switches. There are five individual switches: pitch, yaw, roll, throttle, and vector. These switches will put the aircraft in VSS mode only from attitude mode, which implies that it is impossible to go directly from Direct Mode to VSS Mode, or from FBWD* to VSS. Engagement of individual switches will automatically replace the attitude mode with VSS. For example, when VSS pitch switch is on, the aircraft will fly VSS mode in pitch channel only, roll and yaw in attitude hold, throttle and vectors in FBWD.

7.5 SENSORS

All sensors needed for the VSS are classified into two categories.

*FBWD applies to Pitch, Roll, or Yaw; throttle or vector channels do not have attitude mode.

Category one sensors, which are inexpensive or commonly used in flight controls, are:

Accelerometers, translational and angular

Gyroscopes, rate and attitude

q-sensor

Conventional angle of attack and side slip sensors

Category two sensors, which are expensive or unique sensors, are:

Very low wind velocity sensors

Doppler velocity sensors

Radar altimeter

7.5.1 Low Wind Velocity Sensor

Until recently, Cornell X-22 "LORAS" was the one sensor able to sense low wind velocity. This sensor is sensitive to airplane location and, because of weight and size, presented a very serious airplane installation problem. To overcome this difficulty, the Aeroflex Laboratories sensor was considered. Because of the smaller size it would be easier to find a favorable location for this sensor. This instrument can measure wind velocity in the aircraft speed ranges of 0 - 150, or 0 - 300 knots.

The accuracy of the sensor, ± 0.5 knot, or 0.5 percent, is satisfactory. This accuracy is reported to be unaffected by temperature, humidity, or pressure effects. Because the airstream itself does no mechanical work, the sensor frequency response is high. The sensor itself has a unique design. The airstream flows through a ducted, servo-driven turbine and then across a speed error detector. A closed loop servo continuously synchronizes the turbine speed with the airspeed. A pulse type tachometer is mounted on the turbine shaft to convert the shaft rotation into the voltage signal. The voltage signal is proportional to a wind velocity. The wind velocity direction can be measured from shaft position throughout 360 degrees (synchro or encoder signal can be used).

The outline of this sensor is shown in Figure 7-20.

The weight, size, and power summary is shown in Figure 7-21.

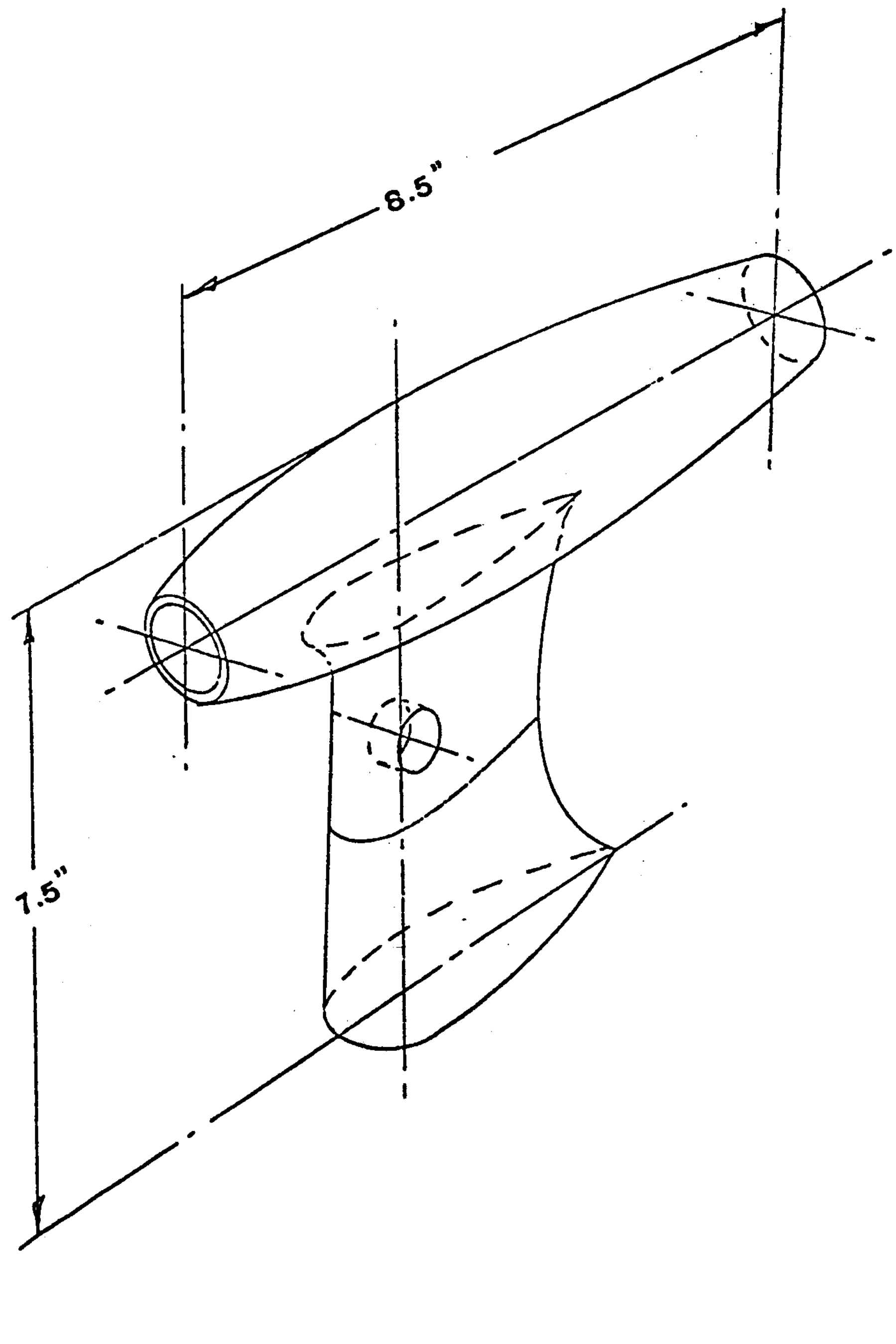


FIGURE 7-20. LOW WIND VELOCITY SENSOR

CATEGORY I		DIMENSION	VOLUME (FT^3)	WEIGHT (LB)	POWER (WATTS)
	$\bar{q}$-sensor		.014 FT^3	3	--
	α & β		.06 FT^3	5	--
	3-axis ang. acc.		.03	2	
	3-axis trns. acc.		.06	1	
	attitude gyro		.06	7	20
CATEGORY II					
Aeroflex Wind Sensors (2)	Velocity sensor	8.5"x2.25" DIA		7	250
	Electronic unit	8"x8"x8"		12	
	Indicator	3.25"x3.25"x6"	.705	3	
Doppler	Antenna, Rec/Tras	18.75"x13.75"x 7.81"	1.156	21.5	160
	Signal Data Conv.	15.68"x 5.03"x7.84"	.356	16.9	
	Console-Indicator	6.5"x5.75"x3"	.963	4	
Radar Altimeter	Ranges of total Systems		.1 - .4	10 - 24	40 - 100

FIGURE 7-21. SENSORS SUMMARY

7.5.2 Doppler Velocity Sensor

Selection of a vendor for the doppler sensor is pending further investigation.

The body mounted-antenna doppler built by the Laboratory For Electronics, Inc. (LFE), is very expensive, and may be a technical risk. None of the LFE units are in production. Only laboratory-type doppler hardware is currently available. Also, there is no information on the doppler measuring error for attitudes above 10 degrees. The doppler equations appear to be mechanized without taking into account pitch and roll angles. This equation mechanization may be in error during a large angle excursion.

The search for a suitable doppler sensor has revealed two other vendors of doppler velocity sensors: Canadian Marconi Company, CMC; and General Precision Laboratories, GPL. Both of these companies have doppler hardware in production which can be supplied at a lower price per unit than the LFE unit. The only problem is that the dopplers are specifically designed for navigational purposes and both have stabilized antennas. These create errors in body velocity measurements for attitude angles 5 degrees or above. To eliminate the error, matrix transformation will be required which means an additional calculation for the airborne computers.

To compare all three dopplers, the same ground-hover condition with zero pitch and roll angles was selected. Figure 7-22 compares all prospective doppler sensors. The GPL doppler is shown to be very attractive from this error consideration.

Further investigation must include the possibility of removing antenna stabilization from the GPL and CMC dopplers. For about 15 degree pitch and 15 degrees roll, all three dopplers should be compared in terms of velocity errors. Any limitation associated with fixed antenna, such as noise, altitude, etc., must be investigated.

7.5.3 Sensor Characteristics

Both category one and category two sensors were investigated and the basic characteristics of these sensors were recorded. The approximate location for the sensors was determined. Figure 7-23 lists characteristics for SAS and VSS sensors and their approximate location on the airplane.

		LFE	CMC	GPL
V_x Velocity error	1σ		.2% + .2 knot	.1% + .1 knot
	2σ	.5% or .25 knot		
V_y Velocity error	1σ		.3% + .3 knot	.1% + .1 knot
	2σ	.5% or .25 knot		
V_z Velocity error	1σ		5% + .2% of $\sqrt{V_x^2+V_y^2}$	.1% + .1 knot
	2σ	.5% or .25 knot		
Hover Resolution		.5 knot	.3 knot	.25 knot
System Time Constant		1 sec	Not defined	1 sec
Radar Altitude	included	no	yes	yes
	error		2% + 2 feet	
Type		3-beam antenna body fixed	3-beam antenna stabilized	3-beam antenna stabilized
TOTAL WEIGHT		55 LBS	42.3 LBS	42.4 LBS
Transceiver		11-1/4"x15-15/16"x17-1/4"	17.5"x16.5"x6"	18.75"x13.75"x7.81"
Data Track		7-5/8"x10-1/8"x12-21/32"	15.2"x7.5"x7.62"	15.68"x5.03"x7.86"
Control		1-7/8"x5-3/4"x2-9/16"	3.75"x5.75"x1.5"	6.5"x5.75"x3"
TOTAL VOLUME		2.27 FT^3	1.45 FT^3	1.37 FT^3
TOTAL POWER		400 WATTS	178 WATTS	160 WATTS

FIGURE 7-22. DOPPLER VELOCITY SENSOR SUMMARY

SENSOR	CHARACTERISTIC	LOCATION
RATE GYRO	Natural Freq. = f_n = 45 cps; damping ζ = .7 Hysteresis = $\pm$ .15%; Resolution = .01 Deg/Sec.	Equipment Bay c.g.
ATTITUDE GYRO	Response: Over 50 cps Error: Vertical = 1/4 Deg.; Pick off = 1/4 Deg.	Nose Compart.
DOPPLER	First order lag: τ = .05 Sec. Deadzone = $\pm$ .25 KN.; 1σ Error = .1% + .1 KN	Nose Compart.
RADAR ALTIMETER	First order lag: τ = .5 sec. Error: Altitude = 1 Ft. + 1%, rate 1 ft/sec. + 3%	Nose Compart. Antennas: Fore and Aft
LINEAR ACCEL. (Force F.B.)	Natural Freq. f_n = 100 cps; damping ζ = .8 Hysteresis = .1%; Resolution = .1%; Deadzone = .1%	Near C.G.
ANGULAR ACCEL. (Force F.B.)	Natural Freq. f_n = 80 cps; Damping = .5 Hysteresis = .1%; Resolution = .1%; Deadzone = .1%	Near C.G.
α & β (Conv.) Angle of Attack and Sideslip	Natural Freq. f_n = 3 cps (For use above 60 KN) Error: 1/4 Deg.	Boom
Aeroflex System (Wind Vel. Meas.)	Response limited by F.B. Servo System Deadzone = .2 KN., Accuracy .5 KN + .5%	Nose (Below Boom)
Gyro Compass GE SR-3	Natural Freq. f_n = 15 cps Null = .15 Deg., Pick off = .233	Aft Safety Pilot Bulk Head

FIGURE 7-23. SAS/VSS SENSORS

7.6 SAS ANALYSIS

7.6.1 N-309 Transfer Functions

Transfer function representations of the N-309 airplane have been computed at four speeds from hover through transitional flight, for a single trajectory. Longitudinal and lateral summaries of these transfer functions are tabulated in Figures 7-24 and 7-25. Numerators shown are for each control quantity specified, with the other control inputs held at zero. Maximum levels of pitch, roll, and yaw reaction control powers agree with 1.5, 2.0, and 1.5 times AGARD 408 requirements respectively.

Values used in calculating the aircraft response to the throttle control inputs are based on data shown for the YJ-85-19 engine shown in Figure 7-27.

Transfer functions representing the N-309 airplane at two extreme speeds in conventional flight are summarized in Figure 7-26.

7.6.2 Compliance With AGARD 408

7.6.2.1 HOVER HANDLING QUALITIES. Minimum damping recommendations for acceptable control of the aircraft rotational motions are specified by AGARD 408 for all speeds from hover to V_{con}. Column 1 in the tabulation below shows these recommended values computed for the N-309 airplane.

	1 Minimum damping specified in AGARD Report 408 $(\text{ft-lb}/\frac{\text{rad}}{\text{sec}})$	2 Gain of SAS rate feedback loop (deg of nozzle disp/$\frac{\text{deg}}{\text{sec}}$)
Pitch	26450	0.306
roll	14500	0.342
yaw	53100	0.897

	Control Input ↓	θ - Motion (DEG) ↓	u - Motion (FT/SEC) ↓	w - Motion (FT/SEC) ↓
$V = 0$ Knots, $\alpha_T = 7.$ DEG, $h = 0$ ft, $\theta_0 = 0$ DEG	Combined Pitch Input $= \delta_H + \delta_Y$ where $\delta_H/\delta_Y = .75$ (δ_H MAX. $= \pm 20$ DEG)	$-.403\left(\frac{s+.0196\pm j.000477}{.0196^2}\right)/\Delta$	$11.52\left(\frac{s+.0196}{.0196}\right)/\Delta$	$-19.89\left(\frac{s-6.08}{6.08}\right)\left(\frac{s+6.10}{6.10}\right)/\Delta$
	Combined Engine Throttles (DEG) (δ_{TC} incremental about 55.5°, δ_{TL} " " 85.5°)	$.118\left(\frac{s+.0196}{.0196}\right)\left(\frac{s-.516}{.516}\right)/\Delta$	$-3.39\left(\frac{s-.513}{.513}\right)\left(\frac{s+.260\pm j.833}{.833^2}\right)/\Delta$	$-21.15\left(\frac{s+.433}{.433}\right)\left(\frac{s-.203\pm j.367}{.420^2}\right)/\Delta$
	Nozzle Vector on Lift Engines δ_w (incremental about Z body axis) (DEG.)	$.827\left(\frac{s+.0253}{.0253}\right)/\Delta$	$.0345\left(\frac{s+.00691}{.00691}\right)\left(\frac{s+.0205}{.0205}\right)/\Delta$	$.0811\left(\frac{s+.00985}{.00985}\right)/\Delta$
		$\Delta = \left(\frac{s+.0253}{.0253}\right)\left(\frac{s+.429}{.429}\right)\left(\frac{s-.204\pm j.366}{.419^2}\right)$		
$V = 50$ Knots, $\alpha_T = 7.$ DEG, $h = 200$ FT, $\theta_0 = 0$ DEG	Combined Pitch Input (as defined above)	$-.599\left(\frac{s+.0234}{.0234}\right)\left(\frac{s+.20}{.20}\right)/\Delta$	$14.41\left(\frac{s+.216}{.216}\right)\left(\frac{s+3.062}{3.062}\right)\left(\frac{s+79.179}{79.179}\right)/\Delta$	$-.102\left(\frac{s+.0117\pm j.0204}{.0235^2}\right)\left(\frac{s+82.499}{82.499}\right)/\Delta$
	Combined Engine Throttles (DEG.) (δ_{TC} incremental about 64°, δ_{TL} " " 85.5°)	$.103\left(\frac{s-.115}{.115}\right)\left(\frac{s+.622}{.622}\right)/\Delta$	$.379\left(\frac{s+.507\pm j.334}{.607^2}\right)\left(\frac{s-10.607}{10.607}\right)/\Delta$	$-2.63\left(\frac{s+.487}{.487}\right)\left(\frac{s-.169\pm j.411}{.444^2}\right)/\Delta$
	Nozzle Vector on Lift Engines δ_w (incremental about 0.3 DEG. aft of Z-axis) (DEG)	$-.878\left(\frac{s+.201}{.201}\right)\left(\frac{s-.00063}{.00063}\right)/\Delta$	$.0018\left(\frac{s+.00119}{.00119}\right)\left(\frac{s+.200\pm j.142}{.245^2}\right)/\Delta$	$.0129\left(\frac{s+.0020}{.0020}\right)\left(\frac{s+.225\pm j6.905}{6.909^2}\right)/\Delta$
		$\Delta = \left(\frac{s+.228}{.228}\right)\left(\frac{s+.466}{.466}\right)\left(\frac{s-.135\pm j.404}{.426^2}\right)$		
$V = 100$ Knots, $\alpha_T = 7.$ DEG, $h = 400$ FT., $\theta_0 = 0$ DEG	Combined Pitch Input (as defined above)	$-1.16\left(\frac{s+.0328}{.0328}\right)\left(\frac{s+.371}{.371}\right)/\Delta$	$2015\left(\frac{s+.749}{.749}\right)\left(\frac{s+.816}{.816}\right)\left(\frac{s+63.66}{63.66}\right)/\Delta$	$-228\left(\frac{s+.0157\pm j.0697}{.0911^2}\right)\left(\frac{s+65.28}{65.28}\right)/\Delta$
	Combined Engine Throttles (DEG) (δ_{TC} incremental about 64°, δ_{TL} " " 65°)	$.0949\left(\frac{s-.224}{.224}\right)\left(\frac{s+.580}{.580}\right)/\Delta$	$.0924\left(\frac{s+.218}{.218}\right)\left(\frac{s+.479}{.479}\right)\left(\frac{s-2.312}{2.312}\right)/\Delta$	$-1.39\left(\frac{s+.536}{.536}\right)\left(\frac{s-.133\pm j.428}{.448^2}\right)/\Delta$
	Nozzle Vector on Lift Engines δ_w (incremental about 1.8 DEG. aft of Z-axis) (DEG.)	$-.791\left(\frac{s+.377}{.377}\right)\left(\frac{s-135.58}{135.58}\right)/\Delta$	$.00197\left(\frac{s+.00142}{.00142}\right)\left(\frac{s+.233}{.233}\right)\left(\frac{s+.549}{.549}\right)/\Delta$	$.0300\left(\frac{s+.00498}{.00498}\right)\left(\frac{s-.545\pm j4.622}{4.654^2}\right)/\Delta$
		$\Delta = \left(\frac{s+.502\pm j.0650}{.506^2}\right)\left(\frac{s-.0977\pm j.387}{.399^2}\right)$		
$V = 150$ Knots, $\alpha_T = 7.$ DEG, $h = 600$ FT., $\theta_0 = 0$ DEG	Combined Pitch Input (as defined above)	$-1.213\left(\frac{s+.0422}{.0422}\right)\left(\frac{s+.538}{.538}\right)/\Delta$	$16.60\left(\frac{s+.491\pm j.592}{.769^2}\right)\left(\frac{s+46.92}{46.92}\right)/\Delta$	$-2.98\left(\frac{s+.0196\pm j.111}{.113^2}\right)\left(\frac{s+47.98}{47.98}\right)/\Delta$
	Combined Engine Throttles (DEG) (δ_{TC} incremental about 64°, δ_{TL} " " 48°)	$.00175\left(\frac{s-.00106}{.00106}\right)\left(\frac{s+7.059}{7.059}\right)/\Delta$	$.887\left(\frac{s+.776\pm j.823}{1.131^2}\right)\left(\frac{s-2.338}{2.338}\right)/\Delta$	$-.635\left(\frac{s+.594}{.594}\right)\left(\frac{s-.102\pm j.429}{.441^2}\right)/\Delta$
	Nozzle Vector on Lift Engines δ_w (incremental about 6.3 DEG. aft of Z-axis) (DEG.)	$-.580\left(\frac{s+.743}{.743}\right)\left(\frac{s-66.01}{66.01}\right)/\Delta$	$.0451\left(\frac{s+.00759}{.00759}\right)\left(\frac{s+.558\pm j1.068}{1.205^2}\right)/\Delta$	$.0322\left(\frac{s+.00908}{.00908}\right)\left(\frac{s-.401\pm j3.541}{3.564^2}\right)/\Delta$
		$\Delta = \left(\frac{s+.0369\pm j.230}{.233^2}\right)\left(\frac{s+.536\pm j1.083}{1.208^2}\right)$		

FIGURE 7-24. HOVER AND TRANSITION LONGITUDINAL TRANSFER FUNCTION SUMMARY OF N-309

	CONTROL INPUT ↑	ψ-MOTION (DEG) ↑	v-MOTION (FT/SEC) ↓	ϕ-MOTION (DEG) ↓
$V = 0$ KNOTS, $\alpha_T = 7.$ DEG, $h = 0$ FT., $\theta_o = 0$ DEG	COMBINED YAW INPUT $= \delta_R + \delta_Z$ WHERE $\delta_Z/\delta_R = .6$, $\delta_{R(MAX)} = \pm 25$ DEG	$\dfrac{400.4\left(\frac{S+.688}{.688}\right)\left(\frac{S-.330\pm j.588}{(.674)^2}\right)}{\Delta}$	$\dfrac{17.8\,S^2}{\Delta}$	$\dfrac{.623\,S^2\left(\frac{S+.0197}{.0197}\right)}{\Delta}$
	COMBINED ROLL INPUT $= \delta_A + \delta_X$ WHERE $\delta_X/\delta_A = 1.0$, $\delta_{A(MAX)} = \pm 15$ DEG	$\dfrac{55.1\left(\frac{S-1.34}{1.34}\right)\left(\frac{S+.678\pm j1.16}{(1.23)^2}\right)}{\Delta}$	$\dfrac{8.61\,S\left(\frac{S+.00288}{.00288}\right)}{\Delta}$	$\dfrac{.301\,S\left(\frac{S+.00288}{.00288}\right)\left(\frac{S+.0197}{.0197}\right)}{\Delta}$
		$\Delta = S\left(\frac{S-.330\pm j.588}{(.674)^2}\right)\left(\frac{S+.00288}{.00288}\right)\left(\frac{S+.689}{.689}\right)$		
$V = 50$ KNOTS, $\alpha_T = 7.$ DEG, $h = 200$ FT., $\theta_o = 0$ DEG	COMBINED YAW INPUT (as defined above)	$\dfrac{17.36\left(\frac{S+.956}{.956}\right)\left(\frac{S-.184\pm j.716}{(.74)^2}\right)}{\Delta}$	$\dfrac{4.92\,S\left(\frac{S-.166}{.166}\right)\left(\frac{S+.626}{.626}\right)\left(\frac{S-138}{138}\right)}{\Delta}$	$\dfrac{45.3\,S\left(\frac{S+.976\pm j3.04}{(3.18)^2}\right)}{\Delta}$
	COMBINED ROLL INPUT (as defined above)	$\dfrac{5.41\left(\frac{S+1.03}{1.03}\right)\left(\frac{S-2.23}{2.23}\right)\left(\frac{S-4.33}{4.33}\right)}{\Delta}$	$\dfrac{5.70\,S\left(\frac{S+.0752}{.0752}\right)\left(\frac{S+3.45}{3.45}\right)}{\Delta}$	$\dfrac{14.5\,S\left(\frac{S+.0653\pm j.331}{(.39)^2}\right)}{\Delta}$
		$\Delta = S\left(\frac{S-.166\pm j.780}{(.797)^2}\right)\left(\frac{S+.0507}{.0507}\right)\left(\frac{S+.952}{.952}\right)$		
$V = 100$ KNOTS, $\alpha_T = 7.$ DEG, $h = 400$ FT., $\theta_o = 0$ DEG	COMBINED YAW INPUT (as defined above)	$\dfrac{7.91\left(\frac{S+1.18}{1.18}\right)\left(\frac{S-.0273\pm j.806}{(.806)^2}\right)}{\Delta}$	$\dfrac{3.58\,S\left(\frac{S-.130}{.130}\right)\left(\frac{S+1.02}{1.02}\right)\left(\frac{S-39.3}{39.3}\right)}{\Delta}$	$\dfrac{41.2\,S\left(\frac{S+.409\pm j2.31}{(2.35)^2}\right)}{\Delta}$
	COMBINED ROLL INPUT (as defined above)	$\dfrac{2.00\left(\frac{S+.921}{.921}\right)\left(\frac{S-2.61}{2.61}\right)\left(\frac{S-9.33}{9.33}\right)}{\Delta}$	$\dfrac{1.84\,S\left(\frac{S+.123}{.123}\right)\left(\frac{S+2.06}{2.06}\right)}{\Delta}$	$\dfrac{10.6\,S\left(\frac{S+.119\pm j.701}{(.71)^2}\right)}{\Delta}$
		$\Delta = S\left(\frac{S+.0107\pm j1.06}{(1.06)^2}\right)\left(\frac{S+.0436}{.0436}\right)\left(\frac{S+1.24}{1.24}\right)$		
$V = 150$ KNOTS, $\alpha_T = 7.$ DEG, $h = 600$ FT., $\theta_o = 0$ DEG	COMBINED YAW INPUT (as defined above) ☆	$\dfrac{-10.47\left(\frac{S+.117\pm j.675}{(.69)^2}\right)\left(\frac{S+1.52}{1.52}\right)}{\Delta}$	$\dfrac{4.71\,S\left(\frac{S-.0419}{.0419}\right)\left(\frac{S+1.59}{1.59}\right)\left(\frac{S+60.2}{60.2}\right)}{\Delta}$	$\dfrac{82.3\,S\left(\frac{S+2.28}{2.28}\right)\left(\frac{S-3.01}{3.01}\right)}{\Delta}$
	COMBINED ROLL INPUT (as defined above) ☆	$\dfrac{-14.76\left(\frac{S+.776}{.776}\right)\left(\frac{S-3.22}{3.22}\right)\left(\frac{S-14.2}{14.2}\right)}{\Delta}$	$\dfrac{-12.9\,S\left(\frac{S+.142}{.142}\right)\left(\frac{S+1.78}{1.78}\right)}{\Delta}$	$\dfrac{-117\,S\left(\frac{S+.172\pm j1.077}{(1.09)^2}\right)}{\Delta}$
		$\Delta = S\left(\frac{S+.0251}{.0251}\right)\left(\frac{S+1.60}{1.60}\right)\left(\frac{S+.158\pm j1.41}{(1.42)^2}\right)$		

☆ REACTION CONTROL SENSITIVITY = 0

FIGURE 7-25. HOVER AND TRANSITION LATERAL-DIRECTIONAL TRANSFER FUNCTION SUMMARY OF N-309

	CONTROL INPUT ↓	LONGITUDINAL DEGREES OF MOTION: θ-MOTION (DEG) ↓	u-MOTION (FT/SEC) ↓	w-MOTION (FT/SEC) ↓
$V = 180$ KNOTS, $\alpha_T =$, $h = 1000$ FT., $\theta_0 =$	COMBINED PITCH INPUT ★ $= \delta_H + \delta_Y$ WHERE $\delta_Y/\delta_H = .75$, $\delta_{H(MAX)} = \pm 20$ DEG.	$\dfrac{-2.66\left(\frac{s+.0526}{.0526}\right)\left(\frac{s+.580}{.580}\right)}{\Delta}$	$\dfrac{31.88\left(\frac{s+.181\pm j.578}{.606^2}\right)\left(\frac{s+55.9}{55.9}\right)}{\Delta}$	$\dfrac{-10.38\left(\frac{s+.0128\pm j.151}{.151^2}\right)\left(\frac{s-56.5}{56.5}\right)}{\Delta}$
	CRUISE ENGINE THROTTLES δT_C INCREMENTAL ABOUT 80° (DEG.)	$\dfrac{.346\left(\frac{s+16.17}{16.17}\right)}{\Delta}$	$\dfrac{.0805\left(\frac{s-.00869}{.00869}\right)\left(\frac{s+.644\pm j1.68}{1.80^2}\right)}{\Delta}$	$\dfrac{-.243s\left(\frac{s+.406}{.406}\right)}{\Delta}$
		$\Delta = \left(\frac{s+.0105\pm j.145}{.145^2}\right)\left(\frac{s+.626\pm j1.69}{1.80^2}\right)$		
$M = 0.75$, $\alpha_T =$, $h = 25000$ FT., $\theta_0 =$	COMBINED PITCH INPUT (as defined above) ★	$\dfrac{-17.56\left(\frac{s+.0261}{.0261}\right)\left(\frac{s+.805}{.805}\right)}{\Delta}$	$\dfrac{379.\left(\frac{s+.593\pm j.552}{.810^2}\right)\left(\frac{s+145.}{145.}\right)}{\Delta}$	$\dfrac{-39.7\left(\frac{s+.0129\pm j.0585}{.0600^2}\right)\left(\frac{s+145.}{145.}\right)}{\Delta}$
	CRUISE ENGINE THROTTLES δT_C INCREMENTAL ABOUT 85° (DEG.)	$\dfrac{.352\left(\frac{s+22.2}{22.2}\right)}{\Delta}$	$\dfrac{.120\left(\frac{s-.00208}{.00208}\right)\left(\frac{s+.836\pm j2.52}{2.65^2}\right)}{\Delta}$	$\dfrac{-.367s\left(\frac{s+.522}{.522}\right)}{\Delta}$
		$\Delta = \left(\frac{s+.0126\pm j.0567}{.0581^2}\right)\left(\frac{s+.835\pm j2.519}{2.65^2}\right)$		

	CONTROL INPUT ↓	LATERAL - DIRECTIONAL DEGREES OF MOTION: ψ-MOTION (DEG) ↓	v-MOTION (FT/SEC) ↓	φ-MOTION (DEG) ↓
$V = 180$ KNOTS, $\alpha_T =$, $h = 1000$ FT., $\theta_0 =$	COMBINED YAW INPUT $= \delta_R + \delta_Z$ WHERE $\delta_Z/\delta_R = .6$, $\delta_R MAX = \pm 25$ DEG. ★	$\dfrac{-21.83\left(\frac{s+1.43}{1.43}\right)\left(\frac{s+.314\pm j.617}{.69^2}\right)}{\Delta}$	$\dfrac{22.6s\left(\frac{s-.0722}{.0722}\right)\left(\frac{s+1.73}{1.73}\right)\left(\frac{s+71.2}{71.2}\right)}{\Delta}$	$\dfrac{1.912\left(\frac{s-.0093}{.0093}\right)\left(\frac{s-5.25}{5.25}\right)\left(\frac{s+2.71}{2.71}\right)}{\Delta}$
	COMBINED ROLL INPUT $= \delta_A + \delta_X$ WHERE $\delta_X/\delta_A = 1.0$, $\delta_A MAX = \pm 15$ DEG ★	$\dfrac{-38.7\left(\frac{s+.481}{.481}\right)\left(\frac{s-3.15}{3.15}\right)\left(\frac{s-28.2}{28.2}\right)}{\Delta}$	$\dfrac{-30.6s\left(\frac{s+.102}{.102}\right)\left(\frac{s+1.43}{1.43}\right)}{\Delta}$	$\dfrac{-3.39\left(\frac{s-.0093}{.0093}\right)\left(\frac{s+.191\pm j1.29}{1.31^2}\right)}{\Delta}$
		$\Delta = \left(\frac{s-.01078}{.01078}\right)\left(\frac{s+1.69}{1.69}\right)\left(\frac{s+.299\pm j1.69}{1.71^2}\right)$		
$M = 0.75$, $\alpha_T =$, $h = 25000$ FT., $\theta_0 =$	COMBINED YAW INPUT (as defined above) ★	$\dfrac{-69.4\left(\frac{s+1.98}{1.98}\right)\left(\frac{s+.168\pm j.525}{.55^2}\right)}{\Delta}$	$\dfrac{35.3s\left(\frac{s-.0102}{.0102}\right)\left(\frac{s+2.12}{2.12}\right)\left(\frac{s+179.}{179.}\right)}{\Delta}$	$\dfrac{36.35\left(\frac{s+3.39}{3.39}\right)\left(\frac{s-.00221}{.00221}\right)\left(\frac{s-3.74}{3.74}\right)}{\Delta}$
	COMBINED ROLL INPUT (as defined above) ★	$\dfrac{-251.8\left(\frac{s+.761}{.761}\right)\left(\frac{s-.444\pm j8.89}{8.90^2}\right)}{\Delta}$	$\dfrac{-158.35s\left(\frac{s+.1745}{.1745}\right)\left(\frac{s+1.87}{1.57}\right)}{\Delta}$	$\dfrac{-13.176\left(\frac{s-.00220}{.00220}\right)\left(\frac{s+.262\pm j2.79}{2.80^2}\right)}{\Delta}$
		$\Delta = \left(\frac{s+.00226}{.00226}\right)\left(\frac{s+.211}{.211}\right)\left(\frac{s+.281\pm j2.97}{2.98^2}\right)$		

★ REACTION CONTROL SENSITIVITY = 0

FIGURE 7-26. CONVENTIONAL FLIGHT TRANSFER FUNCTION SUMMARY OF N-309

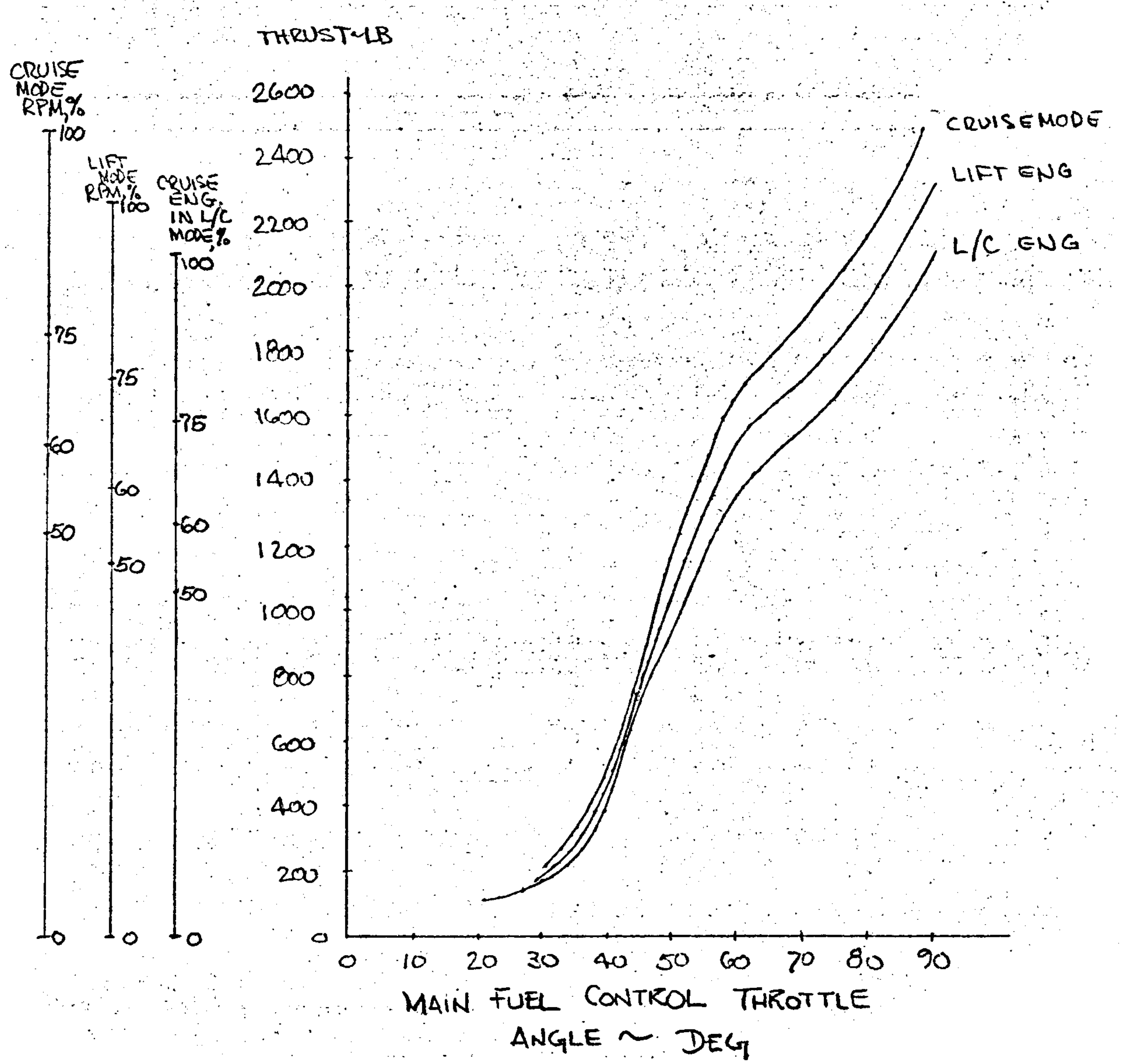


FIGURE 7-27. YJ-85-19 ENGINE THRUST VERSUS MAIN FUEL CONTROL THROTTLE ANGLE INSTALLED, SEA LEVEL, 80° F

Discounting the small natural damping at hover, these AGARD damping recommendations set the minimum SAS gain required at hover in the rate feedback loop. This gain relates rotational velocity of the airplane to displacement from null of the corresponding reaction control nozzle. These values were computed from the following equation, using the pitch axis for example:

$$\delta_{Y}/\dot{\theta} = \frac{B_{(AGARD)}}{(GR)\left(\frac{M_{\delta_Y} \times I_{YY}}{57.3}\right)} = \frac{\text{deg. of nozzle disp.}}{\text{deg/sec}}$$

where:

δ_Y = control disp. of nozzle (degrees)

$B_{(AGARD)}$ = AGARD damping recommendation (ft-lb/$\frac{\text{rad}}{\text{sec}}$)

GR = mechanical ratio (gain) between application of SAS actuator displacement and reaction control nozzle (deg/deg).

M_{δ_Y} = dimensional derivative which defines control sensitivity of reaction control nozzles $\left(\frac{\text{deg/sec}^2}{\text{deg}}\right)$

I_{YY} = angular inertia of the airplane in pitch (Slug -ft^2).

Gains computed as defined above are tabulated in column 2.

AGARD 408 also recommends minimum acceptable dynamic stability criteria applicable from hover to V_{con}. These criteria when combined with the minimum damping requirements establish a sufficient basis for initial mechanization of the SAS control loops.

Figures 7-28 and 7-29 show results of pitch and roll analytical SAS studies at hover for the N-309 airplane. Representative dynamic characteristics for both the power actuators and the SAS actuators have been included. A small amount of attitude signal summed with the basic rate feedback improves the system performance, as illustrated by Figures 7-28 and 7-29, by introducing a small margin of positive static stability. The result satisfied AGARD 408 stability criteria with less of a loss in responsiveness from rate feedback damping. The component of attitude signal may be obtained by integrating the output of a body-rate gyro. This technique requires low frequency filtering to limit the integration of sustained small body rates which occur in steady turns. A SAS loop using rate feedback alone is planned to satisfy AGARD criteria in yaw.

Control power requirements stated in terms of control responsiveness are established by NASA modified AGARD 408. These modified AGARD requirements form the first column of Figure 7-30. Column 2 shows the one-second displacement response to full control power with the SAS off. Dynamics of the pilot's controls have not been included in the response shown in this column. Column 3 is the same as Column 2 except that the dynamics of the control elements connecting the stick to the input to the power actuator has been represented as a 2nd order system with $\zeta = 0.2$ and $\omega_\eta = 31.4$ rad/sec. Column 4 is the same as Column 3 except that the SAS is on. SAS rate feedback gain in all cases is set to conform to AGARD 408 recommended damping. Rate feedback only is used in the yaw SAS, both rate and attitude are used at 5:1 in pitch and 4:1 in roll.

7.6.2.2 <u>TRANSITION HANDLING QUALITIES.</u> Figures 7-31 and 7-32 show results of pitch and roll analytical SAS studies through transition. Acceptable dynamic stability characteristics are attainable using a fixed SAS loop gain at all transitional speeds. This gain, as for the hover study, has been set to provide AGARD 408 minimum damping.

The results of an analytical investigation to determine the response, through the transition range, of a fixed SAS mechanization is shown for the pitch system in Figure 7-33. The one second displacement response is seen to improve with increased forward speed due to the added control effectiveness developed by the aerodynamic surface.

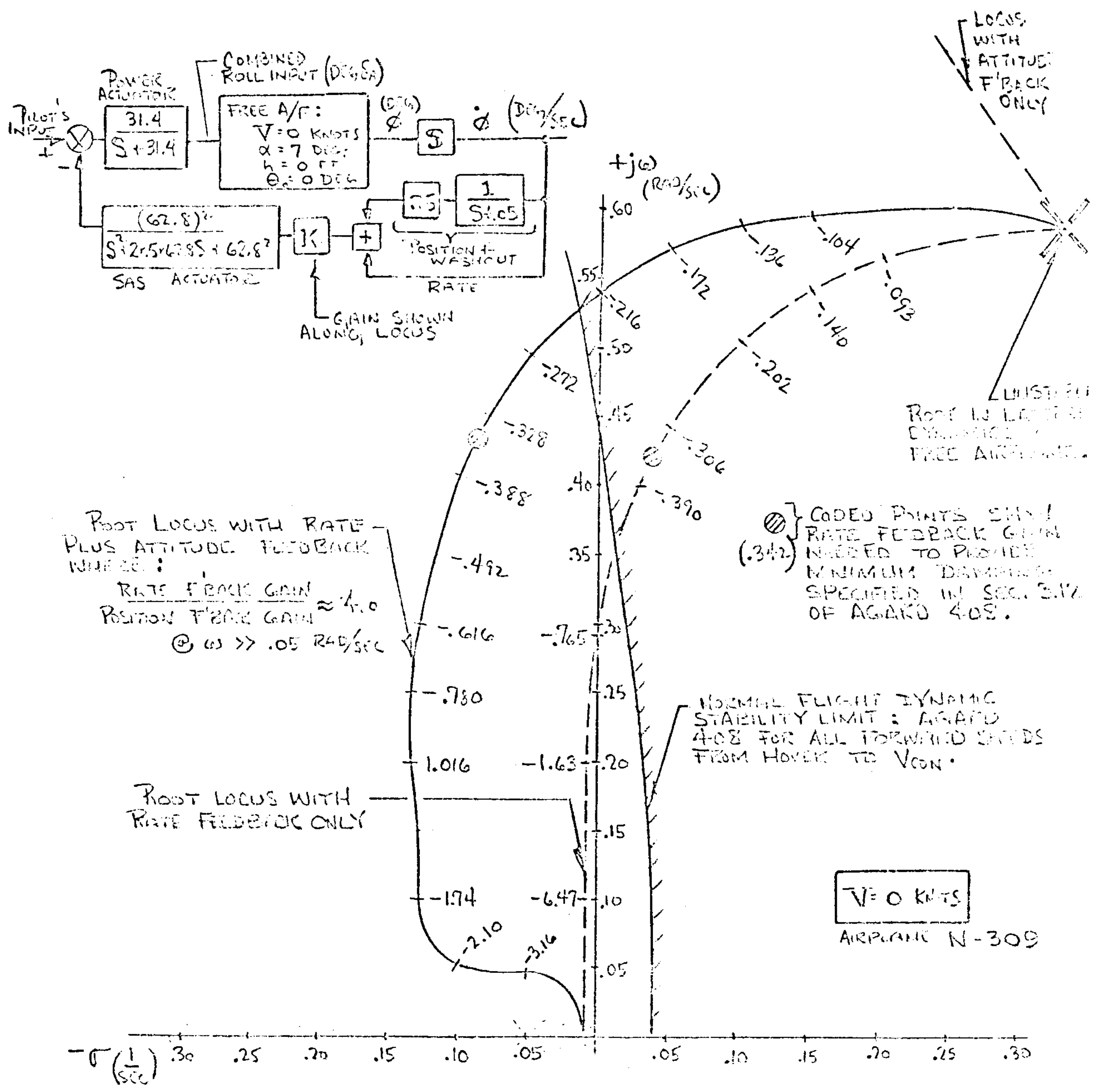


FIGURE 7-28. ROOT-LOCUS STUDY OF ROLL SAS LOOP CLOSURE AT HOVER

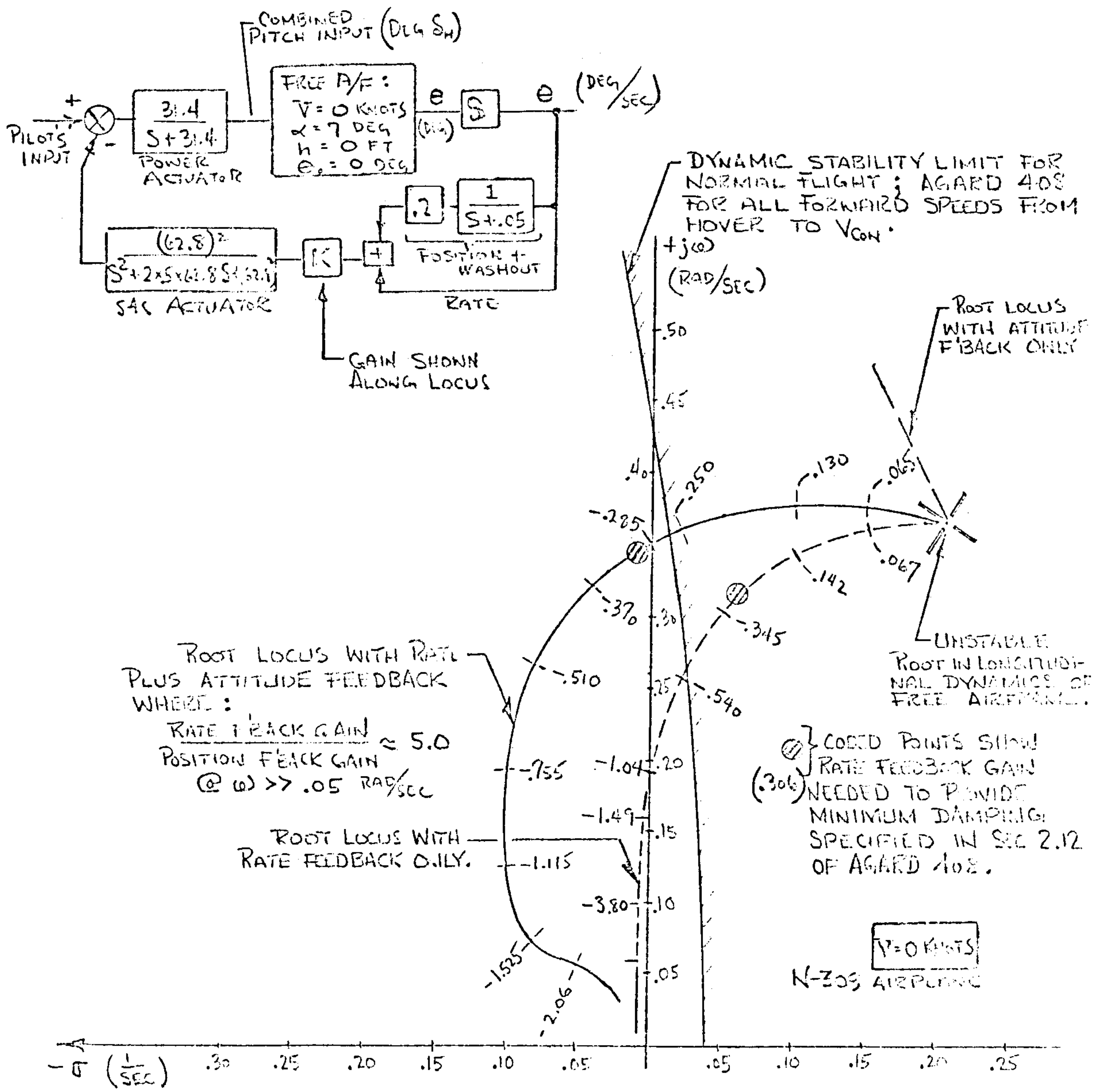


FIGURE 7-29. ROOT-LOCUS STUDY OF LONGITUDINAL SAS LOOP CLOSURE AT HOVER

AIRPLANE RESPONSE TO FULL CONTROL INPUT
(DEGREES IN FIRST SECOND)

	AGARD 408 MODIFIED (NORMAL)	SAS OFF MAX. STEP INPUT (NO CONTROL DYNAMICS)	SAS OFF MAX. STICK INPUT (STEP) ☆	SAS ON MAX. STICK INPUT (STEP) ☆
PITCH	17.0 DEG.	19.8 DEG.	18.06 DEG	15.26 DEG
ROLL	22.6 DEG.	35.62 DEG	32.51 DEG	21.03 DEG
YAW	10.2 DEG.	14.7 DEG	13.39 DEG	10.04 DEG

☆ RESPONSE SHOWN IS CALCULATED WITH THE DYNAMICS OF THE CONTROL ELEMENTS CONNECTING THE STICK TO THE INPUT TO THE POWER ACTUATOR REPRESENTED AS A 2ND-ORDER SYSTEM WITH $\zeta = 0.2$ AND $\omega_n = 31.4$ RAD/SEC

FIGURE 7-30. HOVER RESPONSE OF N-309 AIRPLANE

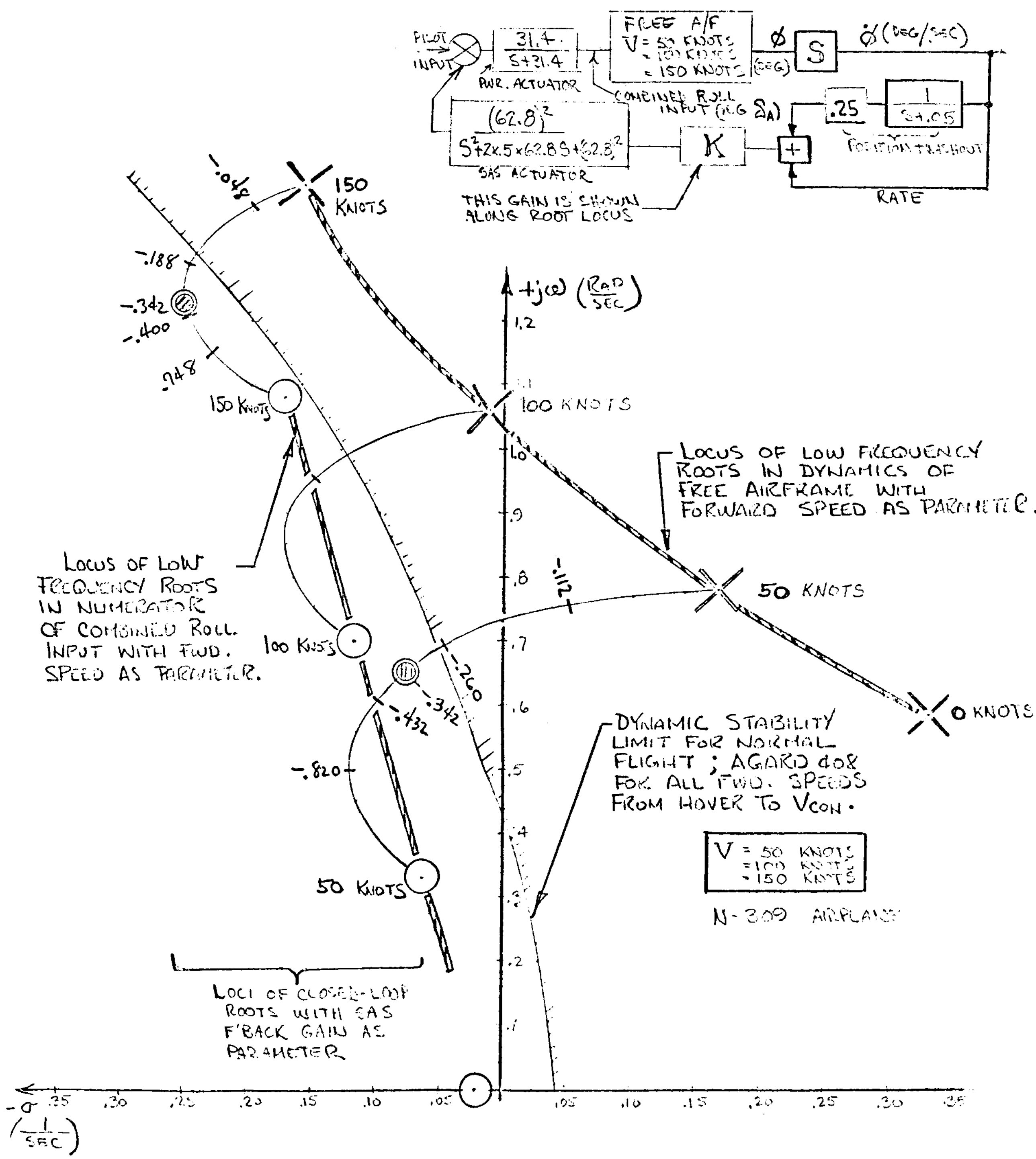


FIGURE 7-31. STUDY OF ROLL SAS LOOP CLOSURE IN TRANSITION

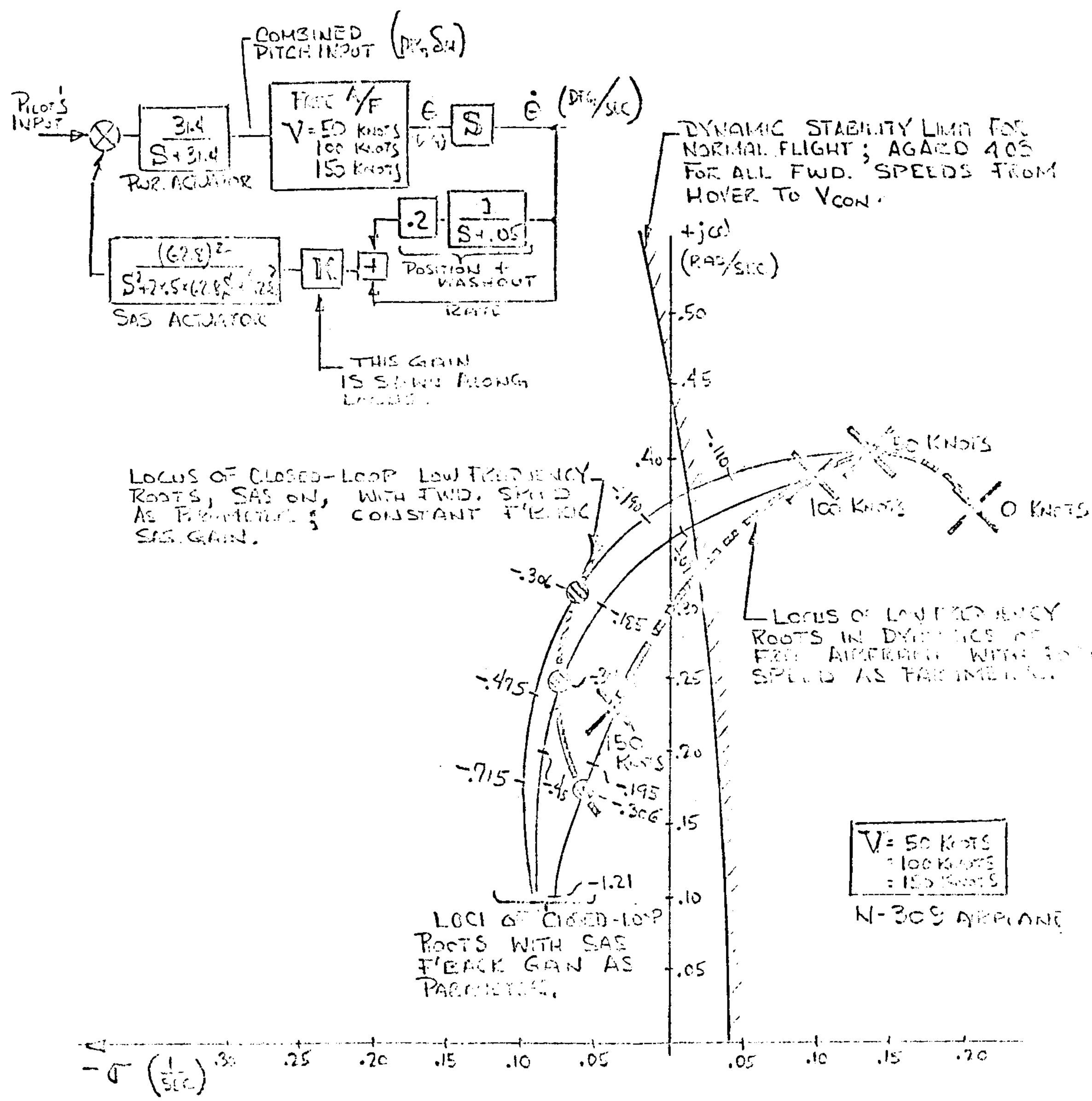


FIGURE 7-32. STUDY OF PITCH SAS LOOP CLOSURE IN TRANSITION

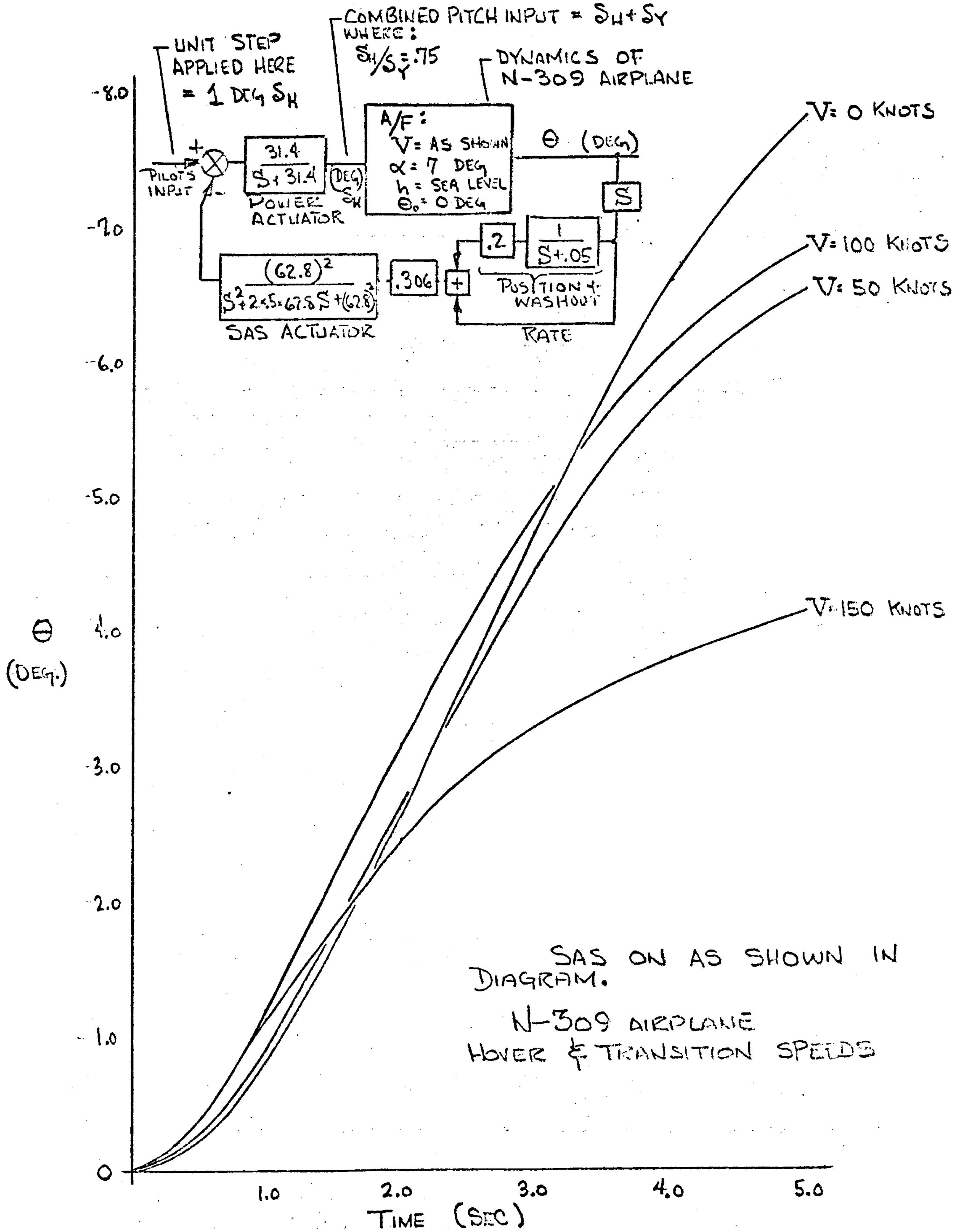


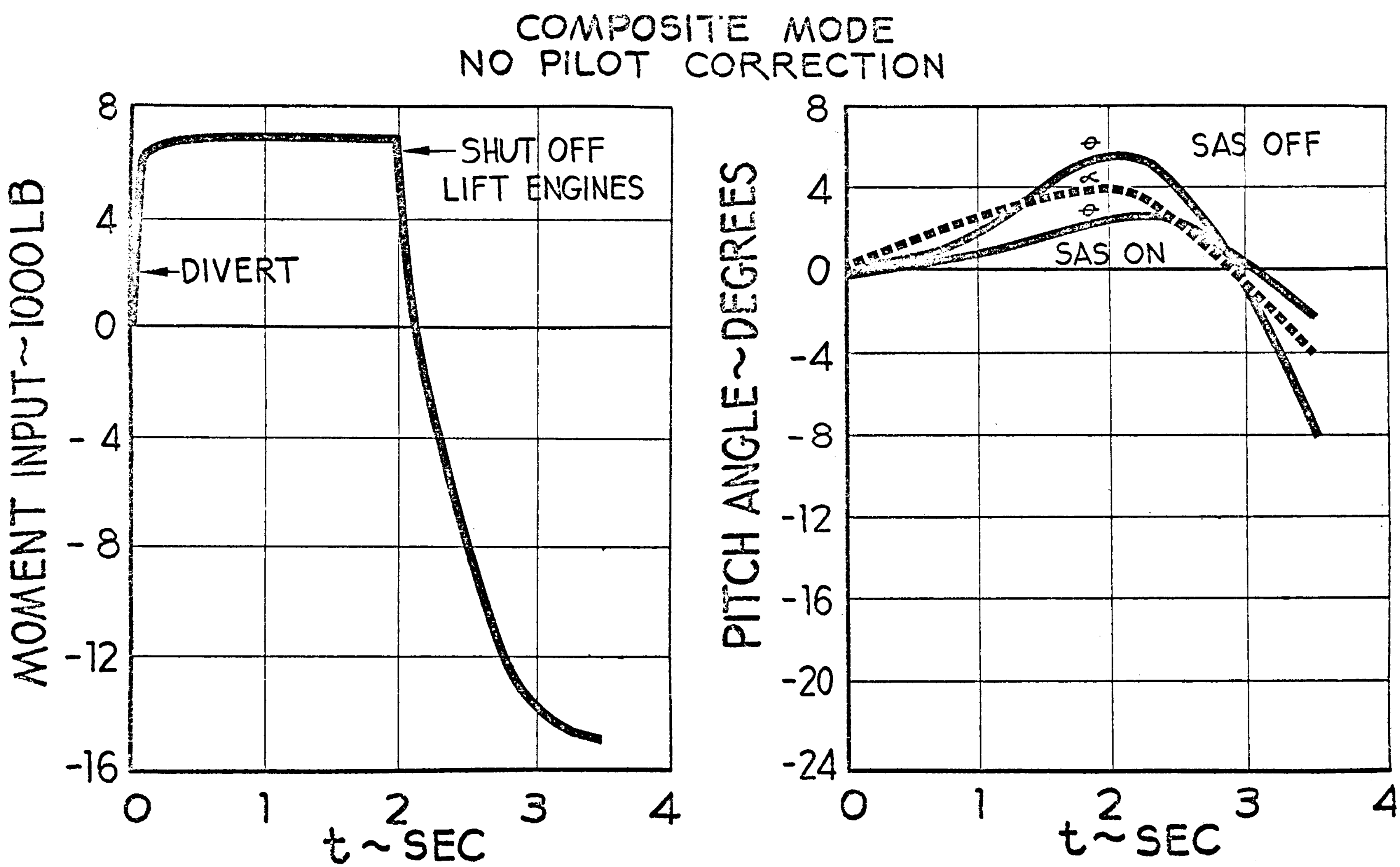
FIGURE 7-33. PITCH TIME RESPONSE TO UNIT STEP INPUT APPLIED BY COMBINED PITCH INPUT

7.6.2.3 CONVERSION DYNAMICS. Conversion dynamics were investigated at a speed of 180 knots. Typical conversion inputs were assumed but to demonstrate the worst case, no pilot stick corrections were applied (see Appendix III for conversion dynamics calculation). The input shown is composed of two parts, the diverter valve change from lift to cruise position, and the shut off of all lift engines (see Figure 7-34). A step input was applied to the diverter valve control first and then, two seconds later, to all lift engine throttles. The time constant of the diverter actuator was 0.05 second; the throttle-engine lag combination was 0.5 second. The airplane response is plotted for SAS OFF and SAS ON. The response with SAS OFF is a pitch attitude of 6 degrees maximum and angle of attack of 4 degrees maximum. With SAS ON the airplane attitude change is less than 3 degrees for the time period shown. No calculations were made for angle of attack with SAS ON.

7.6.3 Longitudinal and Lateral - Directional Handling Qualities (SAS OFF, Conventional Flight, N-309 Aircraft

Minimum acceptable handling qualities in conventional flight are defined by MIL-F-8785. Calculated lateral-directional and longitudinal characteristics for the unaugmented N-309 aircraft in conventional flight are summarized in Figures 7-35 and 7-36. The results show the flying qualities of this configuration, with no augmentation, either exceed or closely approach all minimum requirements of this specification. Handling characteristics defined by the ratio $\delta_{H/g}$ also indicate little probability of impending P.I.O. problems.

A SAS mechanization required to operate only in hover and transition presents many implications. However, many factors such as transients during conversion and uncertainties inherent in preliminary analytical representations must be carefully examined before a decision can confidently resolve this question. Therefore, a more conservative design providing augmentation of reaction controls and aerodynamic controls applicable to hover and transition as well as conventional flight is the recommended preliminary SAS design approach.



10880YZ

FIGURE 7-34. CONVERSION DYNAMICS - N-309

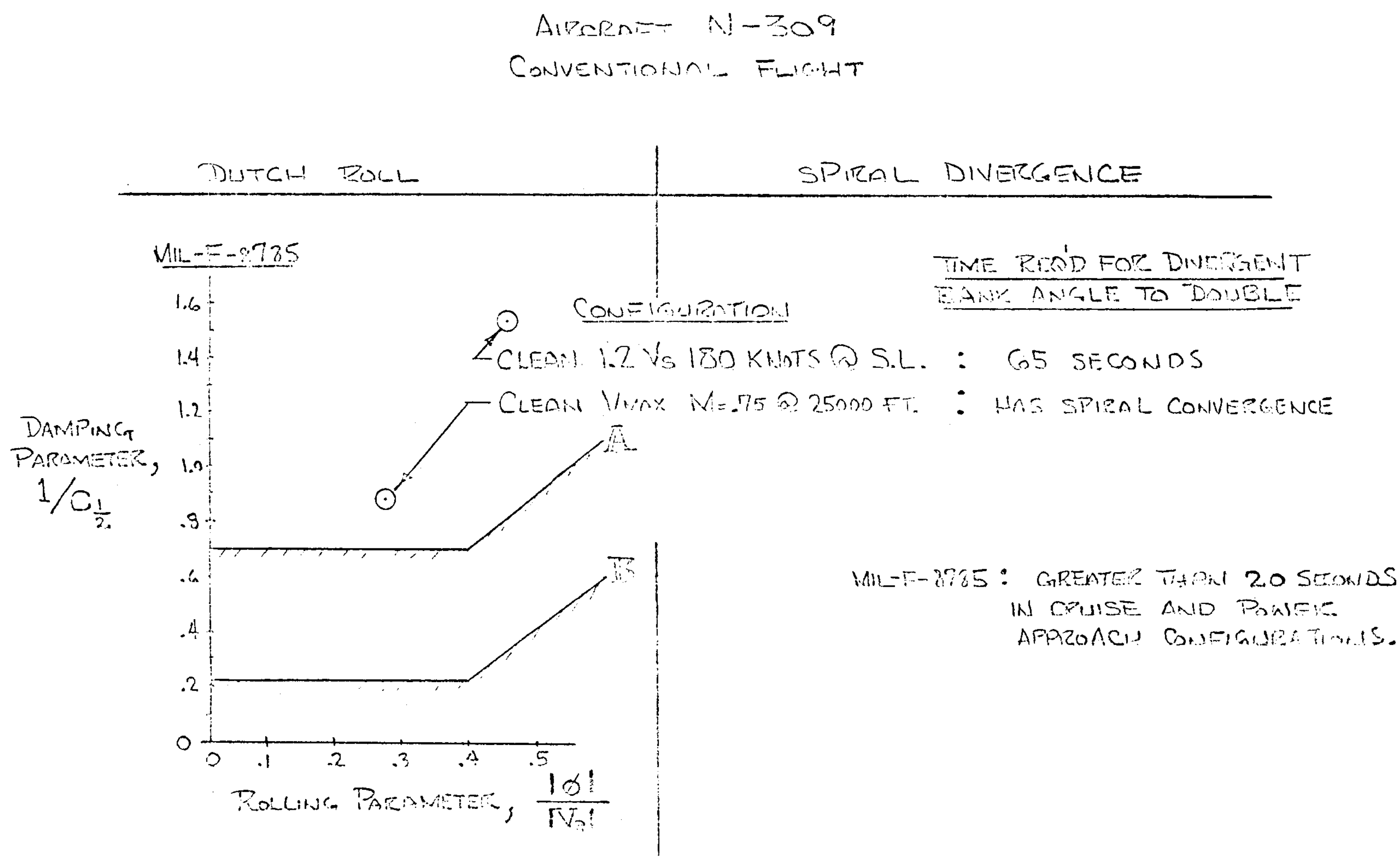
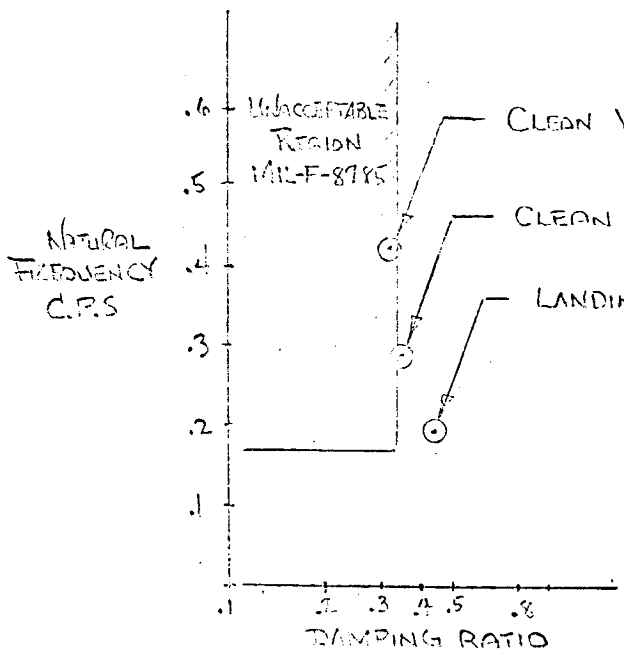


FIGURE 7-35. STABILITY AND CONTROL - LATERAL/DIRECTIONAL - SAS OFF

AIRCRAFT N-309
CONVENTIONAL FLIGHT

SHORT PERIOD	PHUGOID



CONFIGURATION		PHUGOID UNDAMPED PERIOD
CLEAN V_{MAX} M=.75 @ 25000 FT	:	108 SEC.
CLEAN 1.2 V_S 130 KNOTS @ S.L.	:	43 SEC
LANDING 1.3 V_S 150 KNOTS	:	27 SEC

MIL-F-8785 : NO RESTRICTION
IF PHUGOID UNDAMPED
PERIOD > 15 SEC.

FIGURE 7-36. STABILITY AND CONTROL - LONGITUDINAL - SAS OFF

7.6.4 SAS

The Stability Augmentation System (SAS) augments the stability of the basic V/STOL aircraft in pitch, roll, and yaw. The augmentation required is defined by the handling quality criteria of modified AGARD 408 for hover and transition and by MIL-F-8785 for conventional flight. At low speeds the SAS primarily provides rate feedback damping and a compensation network (see Figure 7-37). The SAS gain is held constant from hover to V_{con} and programmed as a function of dynamic pressure between V_{con} and V_{max}. A small amount of attitude signal compensation summed with the rate feedback improves the overall system performance in hover and transition, as illustrated, by introducing a small margin of positive static stability. The result satisfies AGARD 408 stability criteria with less of a loss in responsiveness from rate feedback damping. The component of attitude signal is obtained by integrating the output of a body-rate gyro. This technique requires low frequency filtering to limit the integration of sustained small body rates which occur in steady turns. In the higher speed range between V_{con} and V_{max}, the basic body-rate feedback signal is modified by a lead compensation network programmed also as a function of dynamic pressure.

Three prospective SAS vendors were considered for final selection. All three have acceptable SAS experience. Two of these vendors have experience with dual SAS and one with a triple redundant system.

Figure 7-38 presents the weight, size, and power summary for SAS as reported by vendors. Vendor B proposed a repackaged A7A Stability Augmentation System, Vendor C sent a description of the physical hardware to be used in the SAS. This hardware was based on production programs for the F-111 and F-4 aircrafts.

As Figure 7-38 shows, Vendor B has some weight advantage over Vendor A and Vendor C. However, a small weight advantage is not enough for final selection. Cost, reliability, ground support requirements and interface with VSS are the items for further consideration. For two systems, the total of the recurring and non-recurring costs were considered. Since aircraft dynamics studies for SAS can be shared with VSS, the combined VSS and SAS cost was significant in vendor evaluation. From this consideration the triple redundant system was very attractive.

SAS hardware currently available from production for the L-T-V A-7A has been selected from the Part II survey as best satisfying the SAS equipment of this requirement. The weight shown is the estimated total installed weight and includes all associated plumbing, cables, and brackets. The estimated weight of SAS components alone is 56.6 pounds.

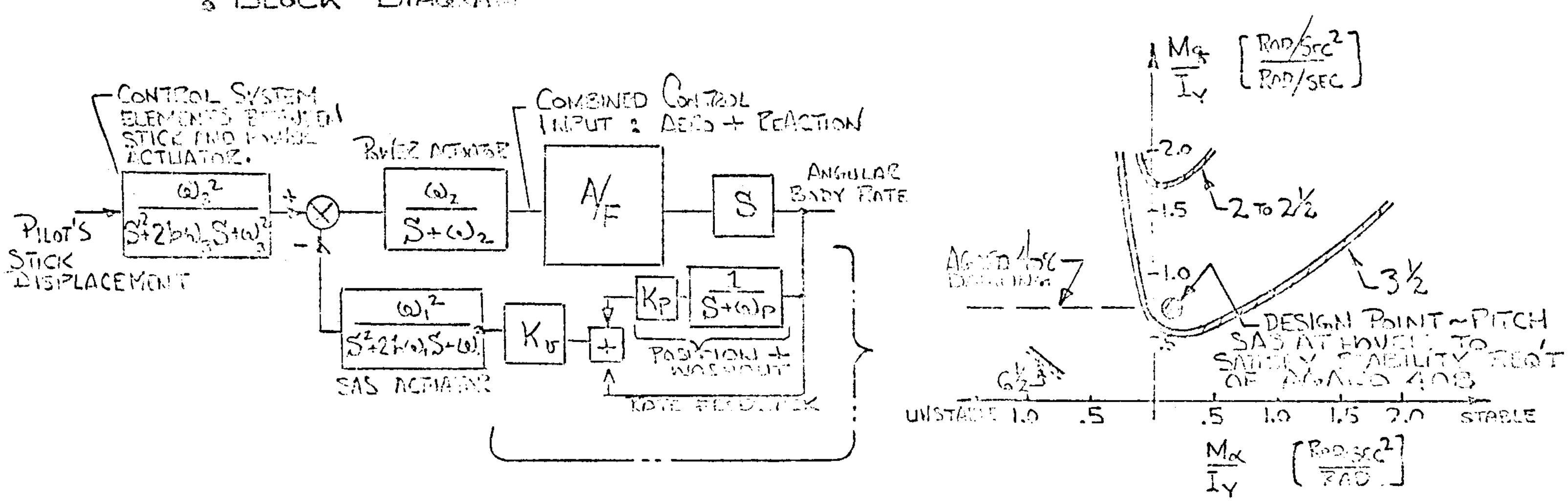
DUAL 3-AXIS SAS WEIGHT SUMMARY	
Controller - Cockpit (2 Required)	6
Electronic Package	16
Servo Amplifier	
Servo Motors	
Gyros - Rate	7
Single Channel Servo Actuators (6 req.)	26
Solenoid shut-off valves (6 req.)	9
Trim mechanism	5
Supports & Brackets	
Plumbing & Fluid	15
Electric Panels, Boxes, Switches, Relays, Wiring	
Pulleys, Sprockets, Chains, Cables & Miscellaneous	34
TOTALS	118

N-309 AIRCRAFT

• CHARACTERISTICS

- WEIGHT : 118 LB.
- POWER : 220 WATTS
- AUTHORITY : FULL FROM TRIM
- RELIABILITY : DUAL REDUNDANT

• BLOCK DIAGRAM



NOTE : PILOT RATINGS SHOWN ABOVE ARE FROM UNPUBLISHED DATA OBTAINED WITH NASA VARIABLE STABILITY HELICOPTER.

FIGURE 7-37. SAS SUMMARY

	DUAL						TRIPLE	
	VENDOR A			VENDOR B			VENDOR C	
	WT LB	DIMENSION	VOLUME (Ft^3)	WT LB	DIMENSION	VOLUME (Ft^3)	WT LB	VOLUME (Ft^3)
Electronic Assembly Unit	25	12"x12"x7"	.7	16	3/8 ATR	.3	44	.4
Front Cockpit Controller	2.5	4"x5.75x3"	.04	3	3"x5"x6"	.052	4.5	.06
Rear Cockpit Controller	2.5	4"x5.75x3"	.04	3	3"x5"x5"	.052	4.5	.06
Rate Gyro Package								
6 gyros	7.5	5"x5"x2.25	.03	5.6	4"x5"x7"	.08		
9 gyros							6	.1
Sub Total Weight (LB)	37.5			30.6			59.0	
Sub Total Volume (Ft^3)			..81			.48		.62
Pitch Servo								
Dual	8	10"x5"x8"	.23	6.25				
Triple							18.5	
Yaw Servo								
Dual	8	10"x5"x8"	.23	6.25				
Triple							18.5	
Roll Servo								
2 Single	10	10"x8"x10"	.46	7				
2 Triple							37.0	
TOTAL WEIGHT (LB)	63.5			50.1			133.0	
TOTAL VOLUME (Ft^3)			1.73					2.3
TOTAL POWER					220 WATTS		*200 WATTS	

*This number was estimated from total power (VSS + SAS)

FIGURE 7-38. SAS WEIGHT, SIZE AND POWER SUMMARY AS REPORTED BY VENDOR

7.7 VSS ANALYTICAL STUDIES

VSS analytical efforts have been directed primarily toward several objectives:

1. To establish VSS Variability Requirements (range of VSS operation).
2. To establish performance criteria for the VSS. The aim is to state performance requirements in terms of aircraft closed-loop dynamic response. This requirement will reflect the dynamic accuracy required of the closed-loop aircraft when performing 5-degree-of-freedom model-following inflight simulations.
3. To analytically assess and evaluate the capability of a basic VSS mechanization approach to satisfy the performance criteria of Item 1 above.
4. To initiate multiloop analysis.

7.7.1 VSS Variability Requirements

The approach undertaken in the study has been to examine VSS variability requirements in terms of bounds placed on variations in the dimensional derivatives appropriate to the VSS airplane-model motion equation. Variability information in this form is needed for specifying VSS computer requirements. Also, these variability studies constitute a preliminary step toward a specification needed to analytically define precisely the types and variation of jet-lift aircraft dynamic characteristics which VSS must accommodate. The realistic approach was taken to specify upper and lower bounds on the dimensional derivatives. Type P-1127 and type CX6 airplanes were selected and for five speeds dimensional derivatives were calculated. Figure 7-39 and Figure 7-40 list the longitudinal derivatives for these two types of VTOL airplanes. Figure 7-39 lists derivatives for speeds V = 0 knots, V = 50 knots, and V = 100 knots. Figure 7-40 lists for two speeds, V = 150 and V = 180 knots. These dimensional derivatives were used to calculate the transfer functions for hover and transition speeds. Both airplane dynamics were compared with N309 dynamics. The result of this comparison is discussed in the following section.

	V=0 KNOTS		V=50 KNOTS		V=100 KNOTS	
Body Axes	P1127	CX-6	P1127	CX-6	P1127	CX-6
Mu	.280	.0070	.287	.0153	.247	.0866
$M_{\dot{w}}$	0.	-.00468	0.	-.00468	0.	-.00468
Mw	.280	.00700	.0663	-.122	-.145	-.187
$M_{\dot{\theta}}$	-.0800	-.0300	-.278	-.0438	-.474	-.0574
M_{θ}	0.	0.	0.	0.	0.	0.
$M\delta_H$	0.	0.	-.662	-.238	-2.62	-.943
$M\delta_Y$	-4.80		-4.00		-3.20	
Zu	0.	0.	-.0203	-.0580	-.04015	-.118
Zw	0.	0.	-.173	-.196	-.343	-.385
$Z_{\dot{\theta}}$	0.	0.	1.45	1.469	2.90	2.921
Z_{θ}	-.0683	0.	-.0683	0.	-.0683	0.
$Z\delta_H$	0.	0.	-.0581	-.0325	-.230	-.128
$Z\delta_T$	-.50		-.50		-.50	
$Z\delta_w$	0.		0.		0.	
Xu	-.0743	-.0268	-.0783	-.0270	-.0690	-.00960
$X_{\dot{w}}$	.000414	.0000218	.000311	.0000218	.000307	.0000218
Xw	0.	0.	-.0310	.0154	-.0621	.0309
$X_{\dot{\theta}}$	0.	0.	-.1925	-.103	-.378	-.204
X_{θ}	-.558	-.562	-.558	-.562	-.558	-.562
$X\delta_H$	0.	0.	.00766	-.00733	.0299	-.0290
$X_{\delta w}$	0.		0.		0.	
$X_{\delta T}$	0.		0.		0.	

FIGURE 7-39. ESTIMATED LONGITUDINAL DERIVATIVES
(VSS RANGE, P1127, CX6 TYPE AIRPLANES)

	V=150 KNOTS		V=180 KNOTS	
Body Axes	P1127	CX-6	P1127	CX-6
Mu	.210	.0750	.167	.0400
$M\dot{w}$	0.	-.00468	0.	-.00468
Mw	-.360	-.337	-.686	-.454
$M_{\dot{\theta}}$	-.633	-.0513	-.731	-.0495
M_{θ}	0.	0.	0.	0.
$M\delta_H$	-5.93	-2.14	-8.51	-3.067
$M\delta_Y$	-2.40		0.	
Zu	-.0739	-.176	-.125	.209
Zw	-.515	-.578	-.616	-.693
$Z\dot{\theta}$	4.38	4.38	5.24	5.23
$Z\theta$	-.0683	-.0196	-.0683	-.0392
$Z\delta_H$	-.522	-.290	-.748	-.418
$Z\delta_T$	-.50		0.	
$Z\delta_w$	0.		0.	
Xu	-.0580	.0269	-.0400	.0648
$X\dot{w}$	.000274	.0000327	.000270	.0000435
Xw	-.0930	.0660	-.094	.102
$X\dot{\theta}$	-.507	-.461	-.599	-.735
$X\theta$	-.558	-.562	-.558	-.561
$X\delta_H$	.0605	-.0270	.0855	-.0242
$X\delta_w$	0.		0.	
$X\delta_T$	0.		.500	

FIGURE 7-40. ESTIMATED LONGITUDINAL DERIVATIVES (VSS RANGE, P1127, CX-6, TYPE AIRPLANES)

7.7.2 VSS Performance Criteria Study

The function of the VSS is to automatically control 5-degree-of-freedom motion response of the basic aircraft in precise phase and amplitude correspondence to electrical signals computed in the onboard reference model. This function obviously implies the cancellation or removal, within a specified bandwidth, of motions which arise from modes which are characteristic of the basic aircraft itself. With all input compensations and with all feedback loops closed, the airplane in VSS mode must be capable of reproducing a uniform motion amplitude response to input voltage commands over a frequency bandwidth which includes the dynamics of all types of aircraft for which the VSS is expected to perform quality simulation.

VSS variability requirements studies have been partially completed in the study to establish preliminary estimates of VSS frequency bandwidth requirements. The approach used has been to specify the required simulation performance of the aircraft in VSS mode on the basis of a range of dynamic characteristics defined by two extreme jet-lift aircraft types. In this manner, the required bandwidth of the basic aircraft in VSS mode can be established by specifying an allowable degree-of-accuracy in the reproduction of this range of dynamic characteristics.

The extreme types chosen for this purpose are the P-1127 type, representing the high response, small, light jet-lift fighter, and the CX-6 type representing a class of heavy, jet-lift transports of limited performance and maneuverability.

The frequency bandwidth of interest is defined by the denominator roots of the lateral and longitudinal transfer functions of these extreme aircraft types. Loci of computed denominator roots, over four flight speeds, for the longitudinal motions are shown by Figure 7-41.

Loci of denominator roots of the N-309 basic aircraft showing the emergence of conventional modes from hover through transition for longitudinal degrees-of-freedom for comparison is shown in Figure 7-42. The lateral-directional degree of freedom is shown in Figure 7-43.

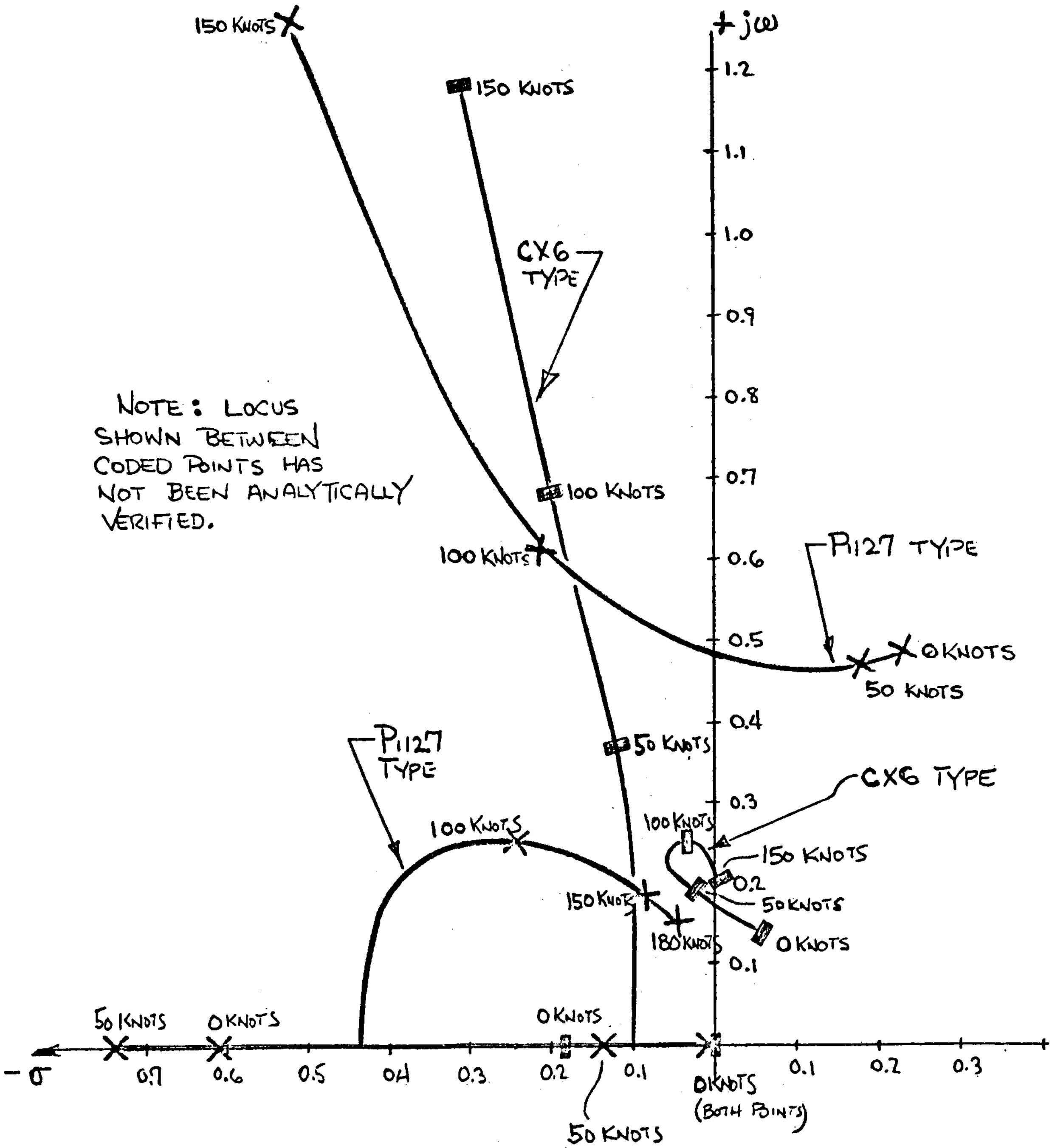


FIGURE 7-41. LOCI OF LONGITUDINAL DENOMINATOR ROOTS OF EXTREME V/STOL AIRCRAFT CHOSEN FOR STUDY OF VSS BANDWIDTH REQUIREMENTS

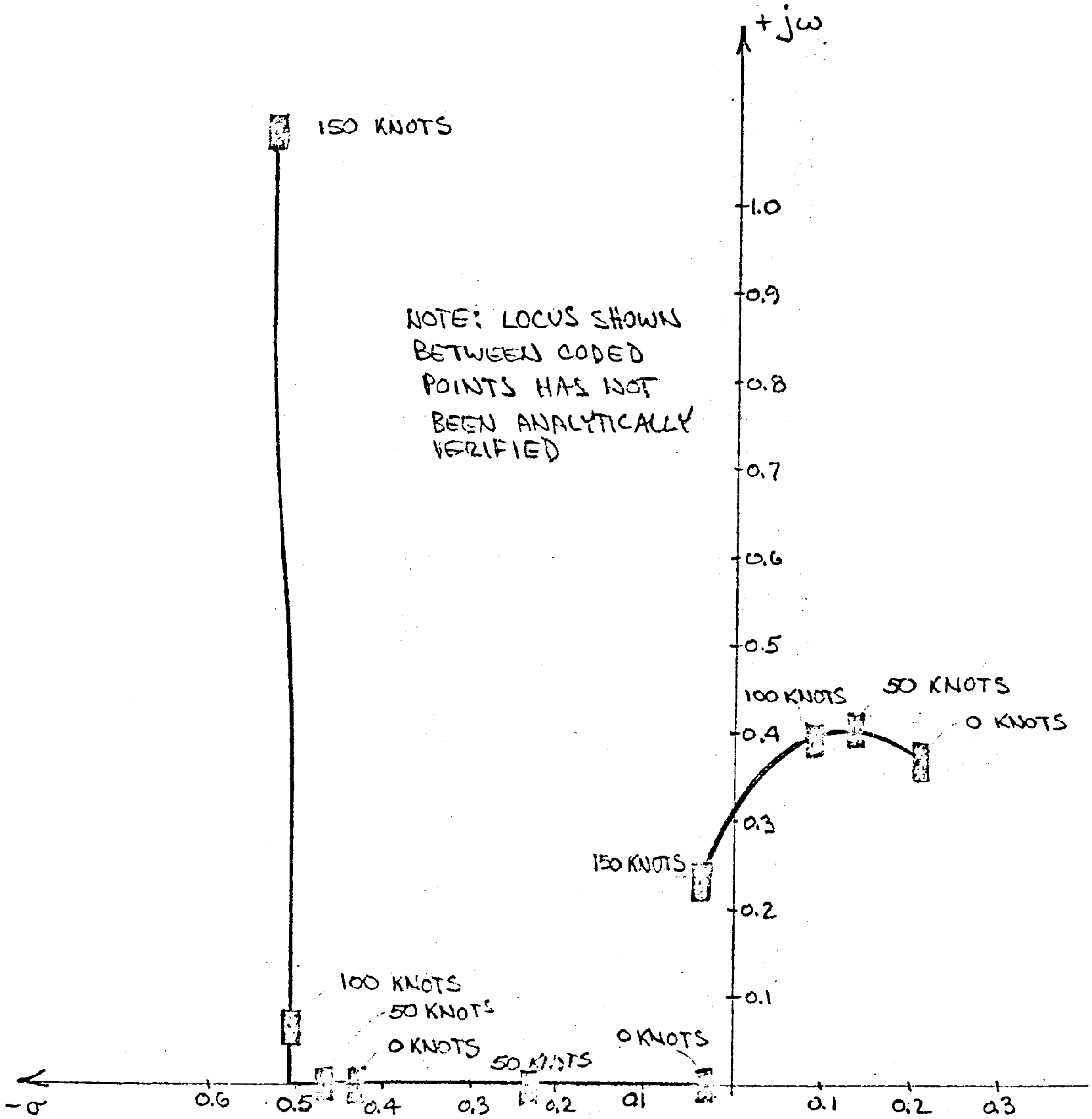


FIGURE 7-42. LONGITUDINAL DENOMINATOR

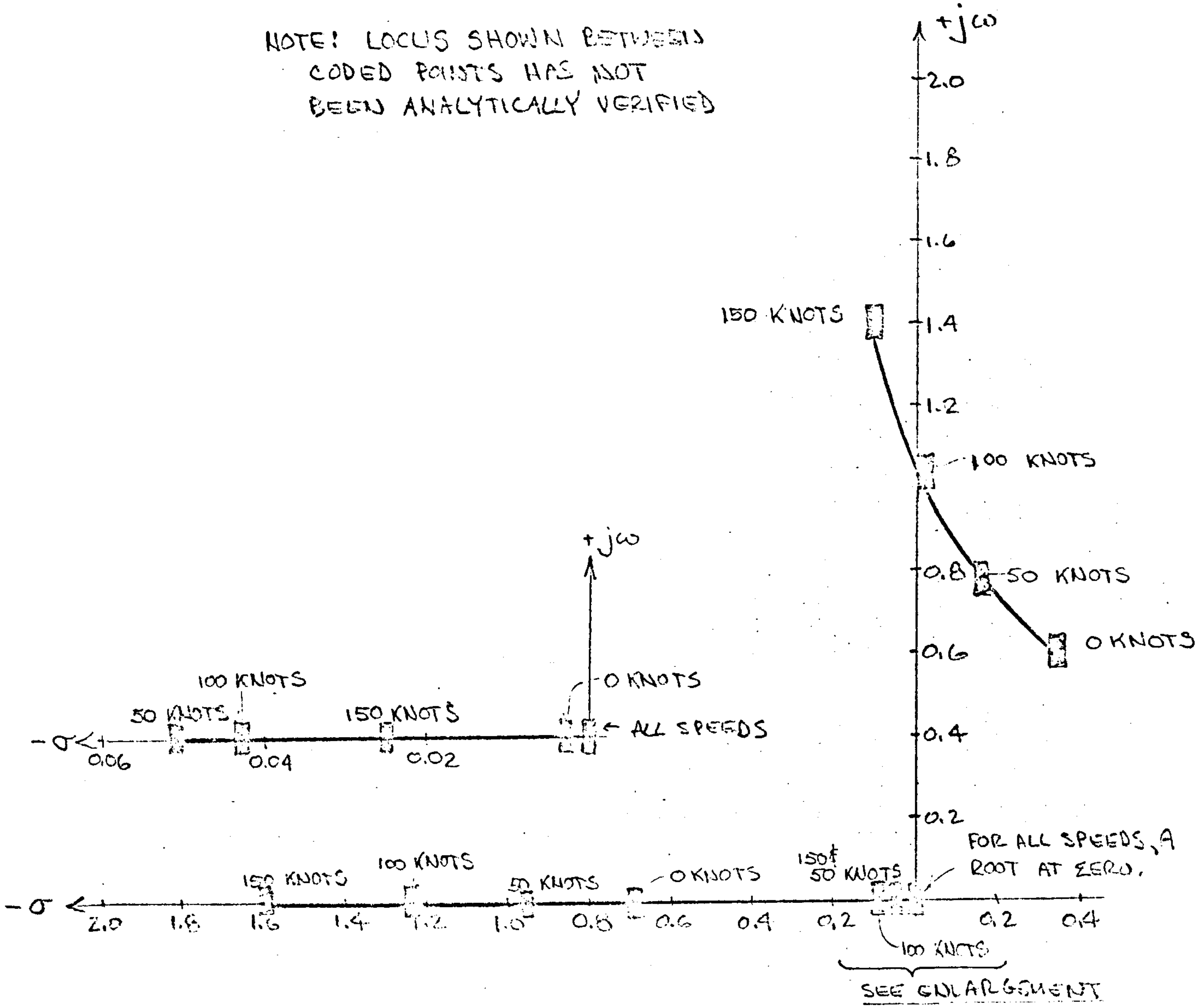


FIGURE 7-43. ROOTS OF LATERAL DENOMINATOR

Using the information from Figure 7-41, it is possible to state the following VSS requirement for simulation in the longitudinal degrees-of-freedom.

Forward Speed (Knots)	Highest Frequency Mode In Extreme Aircraft (rad/sec)	VSS Bandwidth (rad/sec)
0	0.61	3.6
50	0.74	4.4
100	0.71	4.2
150	1.36	8.0

In the above table, the specified bandwidth is chosen to allow not more than 10 degrees of phase lag in the reproduction of the highest frequency mode in the extreme aircraft. A second order upper frequency cutoff characteristics with a damping ratio of 0.5 is needed to adequately define the bandwidth specified.

Further studies are needed to extend the same analysis to the lateral motions.

7.7.3 VSS Mechanization Approach

Achievement of the high bandwidth VSS follower loops required will involve careful application of the best high gain multiloop, cross-coupled synthesis techniques. The VSS mechanization approach presently under study was selected to best satisfy the basic performance criterion for this type of model following closed loop system. The approach is based on the following basic concepts:

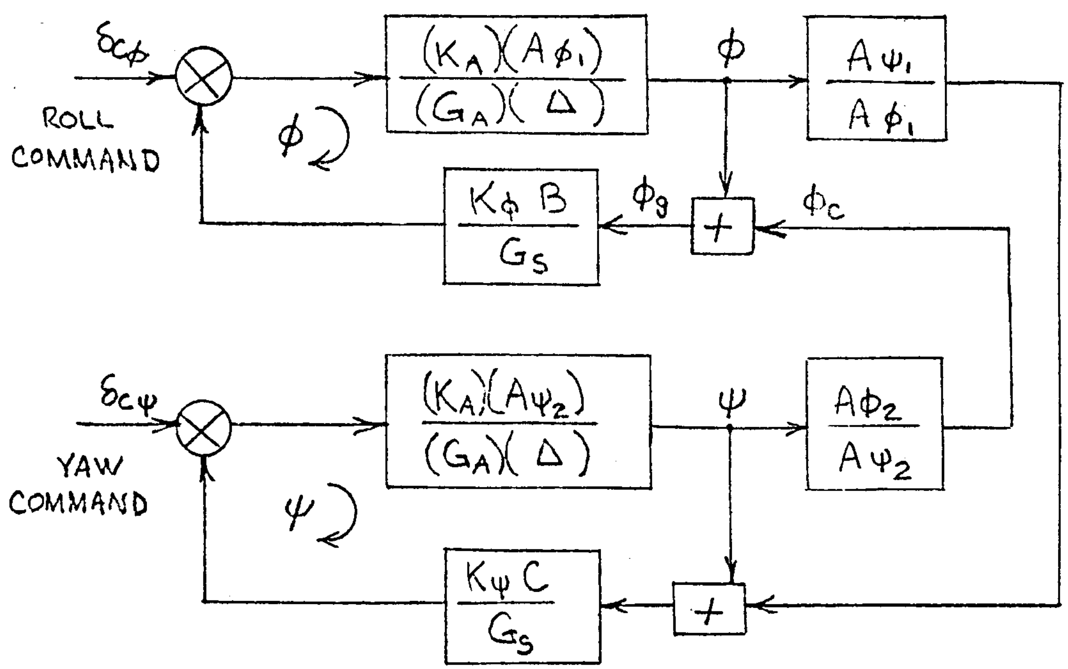
(a) Control or elimination of numerator dynamics in the basic aircraft response will be approached by using feedback loops around the basic airplane, cross coupled as needed between the six output degrees of motion and the five control inputs available, to essentially remove aerodynamic, inertial, and gyroscopic cross coupling between the 5 VSS degrees-of-freedom. This synthesis problem will definitely require closed-loop solutions as multiloop vehicular systems.

(b) The denominator roots which reflect free dynamic modes of the basic aircraft will be drawn by suitable feedback loops with compensation, where possible, to frequencies past the cutoff of the follower bandwidth. Low frequency denominator roots which cannot be modified to this extent will be forced to gain-insensitive locations in stable regions of the left-half S-plane so that cancellation by input compensation can be effectively applied. A technique which introduces feedback zeros by adding together, attitude plus rate and acceleration feedback signals, will be investigated as a means of capturing and holding these low frequency denominator roots in desired regions of the S-plane.

(c) Full advantage will be made of a technique which provides effective noise-free input lead compensation by using a correctly proportioned summation of acceleration plus velocity plus attitude signals in forming the voltage command for the VSS follower loop. The acceleration command signal will augment the basic velocity match at high frequencies. The correctly proportioned addition of displacement in the command will match the low-gain attitude feedback signal and augment the low frequency velocity response as well as add stiffness to the overall system.

7.7.4 Multiloop Analysis Approach

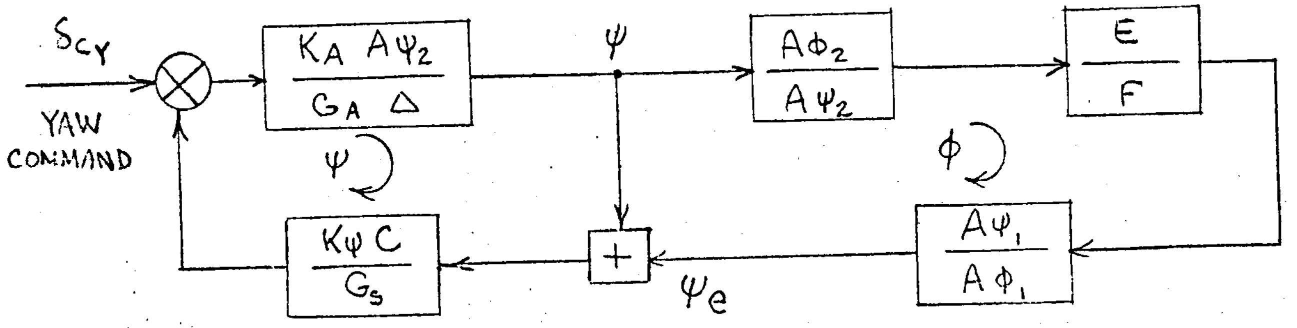
The VSS is a multiloop feedback system. During this study an attempt was made to reduce this system to a single loop, where a conventional analysis approach could be made. The Lateral-Direction system with major feedback loops was analyzed. Figure 7-44 shows multiloop block diagram with roll and yaw commands, and motion crosscoupling. All transfer functions are open loop with all definitions listed. Figure 7-45 shows reduction of feedback loops from three to two. Two approaches are taken. In one approach (top figure) the roll loop was closed with the roll-command set to zero. In the other approach (bottom figure) the yaw loop was closed first, with the yaw command set to zero. Figure 7-46 shows the reduction of two feedback loops into one. The effect of crosscoupling shown as the A_r/B_r transfer function for roll feedback closed first (top figure), and as the A_y/B_y transfer function for yaw feedback closed (below top figure). At the bottom of the figure are examples of the calculations made for hover. A digital computer program was used to find the roots of the A_r/B_r and the A_y/B_y transfer functions. A_r/B_r and A_y/B_y transfer functions in the single loop analysis can be treated as cross coupling "compensation" in the VSS multiloop analysis and its effect on stability, for example, could be found. A simple error analysis then could establish maximum permissible errors in sensors and autopilot equipment, to be used in the final VSS specifications.



$K_A = 31.4$
$A\phi_1$ = FREE A/F NUMER., ROLL COMM, ϕ OUT
$A\psi_1$ = FREE A/F NUMER., ROLL COMM, ψ OUT
$A\phi_2$ = FREE A/F NUMER., YAW COMM, ϕ OUT
$A\psi_2$ = FREE A/F NUMER., YAW COMM, ψ OUT
Δ = FREE A/F DENOMIN

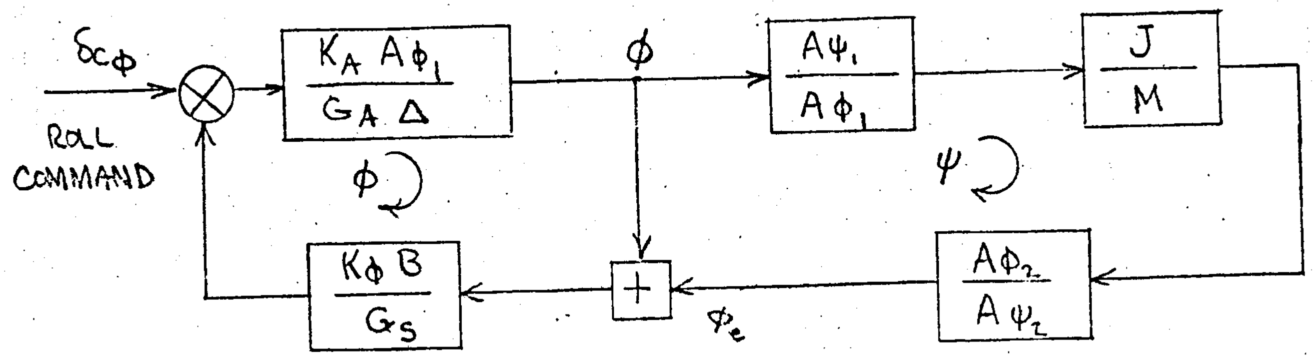
K_ϕ = F.B. GAIN, ROLL LOOP
K_ψ = F.B. GAIN, YAW LOOP
$G_A = (S+31.4)$ — POWER ACT. LAG
$G_S = (S^2+62.8\,S+3943.84)$ — SERVO LAG
B = 3943.84 (F.B. COMPENSAT. - ROLL LOOP)
C = 3943.84 (F.B. COMPENSAT. - YAW LOOP)

FIGURE 7-44. LATERAL-DIRECTIONAL MULTILOOP ANALYSIS



$A\phi_1 = 4.786\ s^3 + .10796\ s^2 + .00027117\ s + 0 \qquad A\phi_2 = .028452\ s^3 + .00055989\ s^2 + 0 + 0$

$A\psi_1 = .020391\ s^3 + .00040125\ s^2 + 0 - .049469 \qquad A\psi_2 = 1.152\ s^3 + .031413\ s^2 + .00017206\ s + .35977$

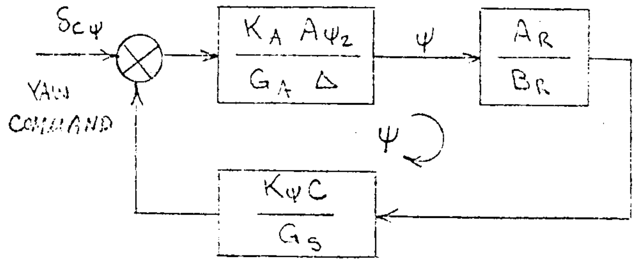


$$\frac{E}{F} = \frac{-K_A K_\phi A\phi_1 B}{\Delta c\phi} \qquad \frac{J}{M} = \frac{-K_A K_\psi A\psi_2 C}{\Delta c\psi}$$

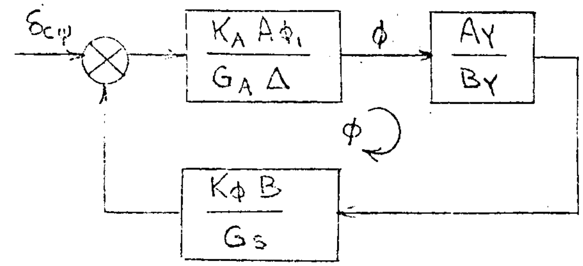
$$\Delta c\phi = G_A \Delta G_S + K_A A\phi_1 K_\phi B \qquad \Delta c\psi = G_A \Delta G_S + K_A A\psi_2 K_\psi C$$

FOR HOVER
$$\begin{cases} \Delta c\phi = .99989\ s^9 + 94.27\ s^8 + 5922.7\ s^7 + 124300\ s^6 + 212420\ s^5 + 67371\ s^4 + 40190\ s^3 + 2050.5\ s^2 + 5.5682\ s \\ \Delta c\psi = .99989\ s^8 + 94.22\ s^7 + 5917.9\ s^6 + 124000\ s^5 + 131730\ s^4 + 5366.8\ s^3 + 33733\ s^2 + 40075\ s \end{cases}$$

FIGURE 7-45. REDUCTION FROM THREE TO TWO LOOPS



$$\frac{A_R}{B_R} = 1 + \frac{(A\phi_2)E(A\psi_1)}{(A\psi_2)F(A\phi_1)}$$



$$\frac{A_Y}{B_Y} = 1 + \frac{(A\psi_1)J(A\phi_2)}{(A\phi_1)M(A\psi_2)}$$

FOR NORMAL

ROLL LOOP CLOSED FIRST

$$\frac{A_R}{B_R} = 1 - \frac{31.4\, s^2(.020291 s^3 + .00040125 s^2 - .049469)(.028452 s + .0005589)(1347.22 s + 404.165)}{(1.152 s^3 + .031412 s^2 + .0017206 s + .35777)(.99989 s^8 + 94.27 s^7 + 5922.7 s^6 + 124200 s^5 + 212420 s^4 + 673715 s^3 + 40190 s^2 + 20505 s + 55632)}$$

YAW LOOP CLOSED FIRST

$$\frac{A_Y}{B_Y} = 1 - \frac{31.4\, s(.020291 s^3 + .00040125 s^2 - .049149)(3537.624 s)(.028452 s + .00055989)}{(4.786 s^3 + .1098 s + .002711)(.99989 s^7 + 94.22 s^6 + 5917.9 s^5 + 124000 s^4 + 131730 s^3 + 5366.8 s^2 + 38733 s + 40075}$$

$$\frac{A_R}{B_R} = \frac{(s-.33 \pm j.588)(s+.035 \pm j.44)(s+.0522)(s+.6378)(s+.002877)(s+1.614)(s+29)(s+31.68 \pm j53.9)}{(s-.33 \pm j.588)(s+.035 \pm j.44)(s+.0522)(s+.6876)(s+.002877)(s+1.614)(s+29)(s+31.68 \pm j53.9)}$$

$$\frac{A_Y}{B_Y} = \frac{(s-.33 \pm j.588)(s+.0197)(s+.6868)(s+.002978)(s+1.093)(s+29.95)(s+31.58 \pm j54)}{(s-.33 \pm j.588)(s+.0197)(s+.6872)(s+.002878)(s+1.093)(s+29.95)(s+31.58 \pm j54)}$$

$$\frac{A_R}{B_R} \cong \frac{(s+.6373)}{(s+.6876)} \cong 1 \;;\; \frac{A_Y}{B_Y} \cong \frac{(s+.6878)}{(s+.6872)} \cong 1 \quad \therefore$$

ZERO CROSS COUPLING BETWEEN ROLL AND YAW (FOR SMALL ANGLES)

FIGURE 7-46. REDUCTION FROM TWO TO SINGLE LOOP

7.8 VSS VENDOR EVALUATION AND FINAL WEIGHT SUMMARY

Several vendors have expressed the desire to build the autopilot and the associated equipment, but only two vendors have responded to the VSS data package with useful information. Most autopilot information was taken from experience with airplanes such as the F-111B, A-7A, XV-4B, and F-4.

Without a detail autopilot specification, or without knowledge of airframe dynamics and more definite control system descriptions, the vendors could only respond on the basis of experience gained with other airplanes, and to make a gross estimation of the autopilot.

General Electric anticipates that most of the autopilot functions can be implemented in the digital computer. The main function then is to provide the actual servo amplifier for the various actuators and the interfacing of the cockpit control force and position sensors with the SAS and VSS functions.

Lear Siegler anticipates more complex autopilot functions with all safety switches. In view of this approach, the Lear autopilot probably would require more power.

Both vendors felt that any definite design statements could not be answered prior to the first part of the design phase.

General Electric proposed the parallel hydraulic servo actuator specifically designed for the F-111B application. This actuator has engagement features that can be directly used in switching techniques between the safety pilot and the evaluation pilot. The safety features designed for F-111B carrier landing are also usable for the NASA V/STOL airplane.

Lear Siegler proposed three choices of approach to parallel actuators, all of which are "off-the-shelf" items. One was a hydraulic actuator with a three way valve. two others were electro-mechanical servos. Recommendation of a specific actuator depends on additional information concerning response and load requirements.

The summary of autopilot and associated equipment (VSS without computer, radar altimeter, doppler, wind velocity sensor, and gyros) is presented in Figure 47 as proposed by these two vendors.

Figure 47 presents VSS and SAS subsystem summary. The data was taken from vendor responses. Weight, volume and power are presented in the ranges to indicate the variation existing between equipment and vendors. The table below lists VSS weight intended for N-309 or MOD. T-39 installation.

VARIABLE STABILITY SYSTEM

DESCRIPTION	Equipment LB	Installation LB	Total LB
Pitot Boom and Mount	14		14
q Sensor	3		3
Angle of Attack and Sideslip Sensors	5		5
Aeroflex Sensors	7		7
Aeroflex Unit and Indicator	15		15
3 Axis Accelerometers - Angular	2		2
- Translation	1		1
Attitude Gyro	7		7
Airborne Computer	74		74
Autopilot Unit	18		18
Cockpit Controllers	8		8
Transducers	16		16
Parallel Actuators	21		21
Radar Altimeter - Rec/Xmitter	10		10
- Antennas	2		2
Doppler Radar - R-T Unit and Antenna	21		21
- Signal Converter	17		17
- Cockpit Controller (1)	4		4
- Mounting Rack		2	2
Provisions For Future Units	86	43	129
Installation - Electronic Units		62	62
- Servo/Mechanical		62	62
Totals	331	169	
Total Variable Stability System			500

VENDOR	LEAR SIEGLER DIMENSIONS Each	LEAR SIEGLER VOLUME Ft^3	LEAR SIEGLER WEIGHT Lbs	GENERAL ELECTRIC DIMENSIONS Each	GENERAL ELECTRIC VOLUME	GENERAL ELECTRIC WEIGHT Lbs
Autopilot Unit	3/8 ATR	.7	18		.4	22
VSS Cockpit Controller	6"x5"x8"	.14	8			6
Front and Rear Emergency Disconnect Switch			.2			
Pitch Position Transducer (2)	2"x2" DIA	.01	.86			.4
Yaw Position Transducer (1)	2"x2" DIA	–	.43			.2
Roll PositionTransducer (1)	2"x2" DIA	–	.43			.2
Pitch/Roll Force Transducer(2)	6"x2" DIA	.02	10			3
Yaw Force Transducer (2)	6"x1.5" DIA	.01	4			4
Throttle Position Transducer (2)	2"x2" DIA	-	.86			.4
Vector Position Transducer (1)	2"x2" DIA	-	.43			.4
Parallel Actuators Hydraulic (5)			21.3	4"x4"x12"		39
TOTAL WEIGHT			64.5			75.2
TOTAL VOLUME (Actuators Excluded)	.72FT^3					
TOTAL POWER	340 WATTS			130 WATTS		

FIGURE 47 AUTOPILOT AND ASSOCIATED EQUIPMENT SUMMARY (UNINSTALLED)
(VSS Summary, not including Radar Altimeter, Doppler, Wing Velocity Sensors and Gyros)

	WEIGHT (LBS)	VOLUME (Ft^3)	POWER (WATTS)
COMPUTER	57 to 74	1.2 to 2.5	170 to 440
VSS (excluding Flight Sensors, Computer and Throttle and Vector Controls)	65 to 75	1.27 to 1.5	130 to 340
SAS (includes Rate, Gyro Package)	50 to 173	1.73 to 3.5	200 to 220
FLIGHT SENSORS (Category I)	16 to 20	.2 to .23	18 to 22
FLIGHT SENSORS (Category II)	74 to 101	2.18 to 3.38	450 to 750
VSS and SAS TOTAL	262 to 443	6.58 to 11.1	968 to 1770
VECTOR CONTROLS THROTTLE CONTROLS	15	.3	180
TOTALS	277 to 458	6.88 to 11.4	1,150 to 1,950

FIGURE 48. VSS AND SAS SUBSYSTEM SUMMARY
Showing Estimates of Values Per Airplane

7.9 CONCLUSIONS AND RECOMMENDATIONS

(1) Vendor's response to the VSS Data Package was favorable.

(2) Vendors, in general, agree with proposed concepts.

(3) Airborne digital computer was selected for VSS functions.

(4) Triple redundant system for SAS is a serious competitor to the dual system.

(5) Further VSS analytical studies are needed.

(6) SAS analysis should be extended.

(7) SAS and VSS study should be combined to investigate the possibility of leaving SAS loop closed in VSS operation.

(8) Further study is needed for final digital computer selection.

(9) The selection of doppler velocity sensor is pending further investigation.

(10) The need for radar altimeter depends on the type of doppler chosen for VSS.

(11) Radar altimeter selection should be based on descent velocity error, rather than an altitude error.

(12) Extreme care is necessary for accurate simulation using the VSSD mode.

APPENDIX 7-I

HOVER REQUIREMENTS

LONGITUDINAL RESPONSE & DAMPING CHARACTERISTIC HOVER

Following holds for any allowable C.G. and the most critical combination of weight and moment of enertia.

Response for first inch of control displacement from trim should be equal to or greater than the response per inch of remaining travel.

Longitudinal Response

	Response for full control input (degs in first sec)	Response for first inch of control displacement (degs in first sec)	Damping lb ft / rad/sec
Normal conditions	$\frac{450}{(W+1,000)^{1/3}}$	$\frac{112.5}{(W+1,000)^{1/3}}$	$15(I_y)^{0.7}$
After single failure in PCS or SAS	$\frac{270}{(W+1,000)^{1/3}}$	$\frac{67.5}{(W+1,000)^{1/3}}$	$8(I_y)^{0.7}$

Directional Response

	(Degs in 1st Sec) Full Control Input	(Degs in 1st Sec) First Inch of control	Damping lb ft / rad/sec
Normal conditions	$\frac{270}{(W+1,000)^{1/3}}$	$\frac{90}{(W+1,000)^{1/3}}$	$27\ (I_z)^{0.7}$
After a single failure in a PCS or SAS	$\frac{270}{(W+1,000)^{1/3}}$	$\frac{90}{(W+1,000)^{1/3}}$	$14\ (I_z)^{0.7}$

Lateral Response

	(Degs in 1st Sec) Full Control	(Degs in 1st Sec) Response for 1st inch of control	Damping (lb ft / rad/sec)
Normal Conditions	$\frac{600}{(W+1,000)}^{1/3}$	$\frac{200}{(W+1,000)}^{1/3}$	$25\ (I_X)^{0.7}$
After a single failure in a PCS or SAS	$\frac{600}{(W+1,000)}^{1/3}$	$\frac{200}{(W+1,000)}^{1/3}$	$18\ (I_X)^{0.7}$

Above hover requirements are specified in order to ensure satisfactory control power and sensitivity for maneuvering and to minimize the effects of external disturbances.

TRANSITION CHARACTERISTICS

1. With aircraft trimmed in hovering flight, it should be possible to accelerate rapidly and safely to VCON at approximately constant altitude. From trimmed steady level unaccelerated flight at VCON it should be possible to decelerate rapidly and safely at approximately constant altitude to stop and hover.
2. In accelerating and decelerating transition, control and response characteristics of the aircraft should vary in a smooth manner throughout the transition maneuver.
3. At all forward speeds and loadings between hover and VCON longitudinal oscillation with controls fixed should exhibit damping characteristic not less than that given by the normal flight curve in Figure 7I-1. To facilitate analysis of aircraft dynamics, equivalent boundaries have been plotted in the complex plane (See Figure 7I-2).
4. Lateral/directional oscillations should exhibit characteristics as indicated in Figures 7I-1 and 7I-2. Spiral stability should be positive for all normal flight conditions up to VCON.

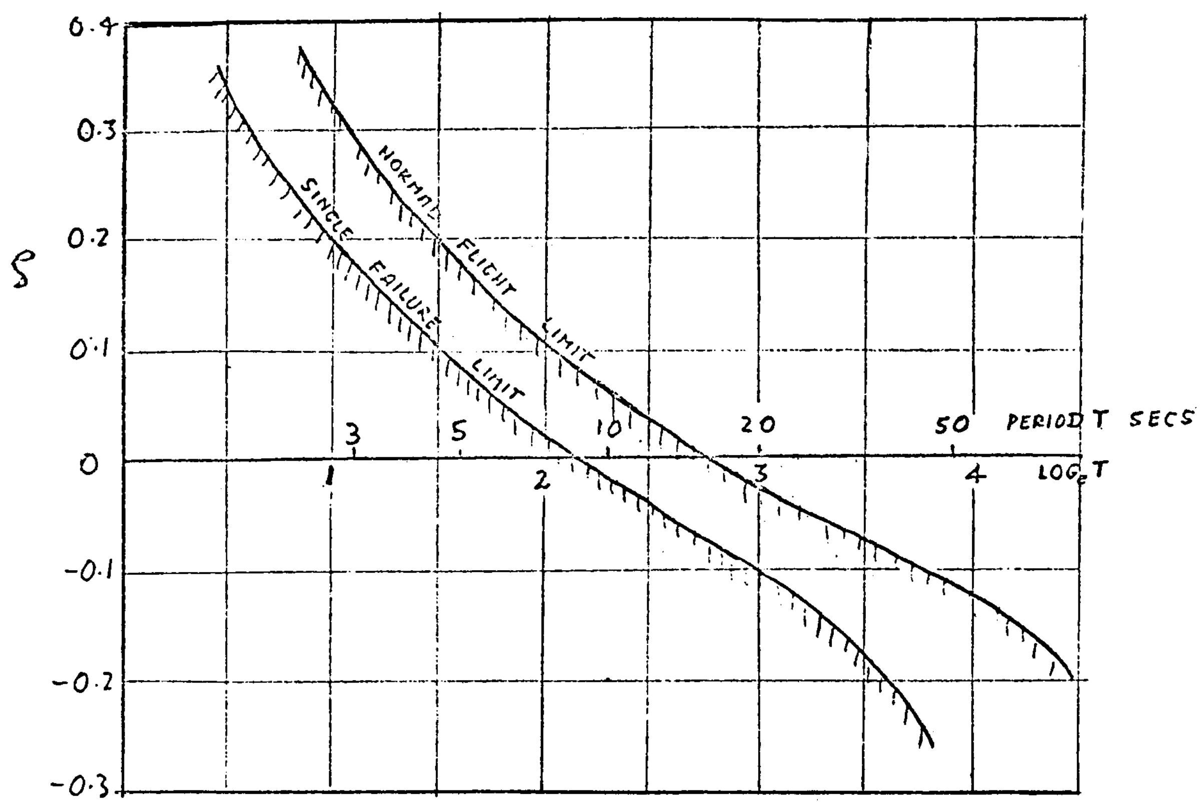


FIGURE 7I-1. AIRFRAME DYNAMIC STABILITY SPECIFICATION (CONTROLS FIXED) TRANSITION REGION (All Forward Speeds From Hover to VCON).

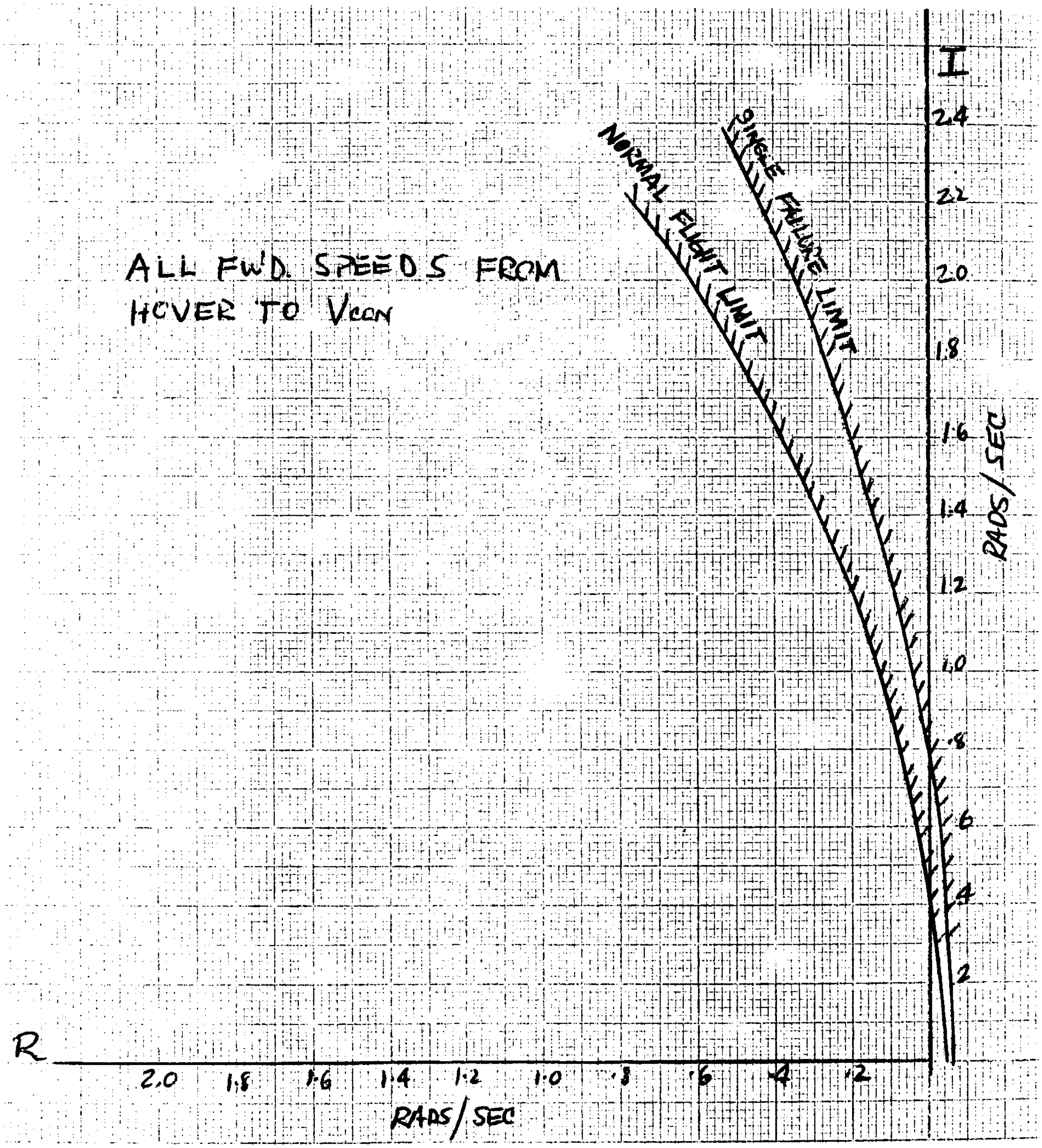


FIGURE 7I-2. AIRFRAME DYNAMIC STABILITY SPECIFICATION (CONTROL FIXED)
TRANSITION REGION (FREQ. DOMAIN)

GENERAL REQUIREMENTS
HOVER & TRANSITION

The following requirements apply to hovering and transition flight.

1. After a sudden rearwards longitudinal control input, sufficient to generate a 0.2 rad/sec pitch rate with 3 seconds, the time history of angular velocity should become concave downward within 2 seconds following the start of the maneuver and remain concave downward until the attainment of maximum angular velocity. (See Figure 7I-3).

2. The instability of the basic aircraft should not be so great that during any longitudinal maneuver within the design flight envelope the input of the stability augmenter together with the pilot's input at any time leaves less that 50% of the nominal longitudinal control moment for recovery.

3. Basic lateral and directional instability should not be so great that in the sideslip or side translations specified, the input of the stability augmenter system together with the pilot's input, at any time leaves less than 50% of the nominal directional and lateral control moment for recovery. (Specified sideslip or side translations are those produced by full directional control with stability augmenter system in operation).

CONVENTIONAL FLIGHT
CHARACTERISTICS

1. Longitudinal short period oscillatory characteristics should conform to the specification boundaries of Figure 7I-4.

2. Lateral/directional oscillatory characteristics should conform to the specification boundaries of Figure 7I-5.

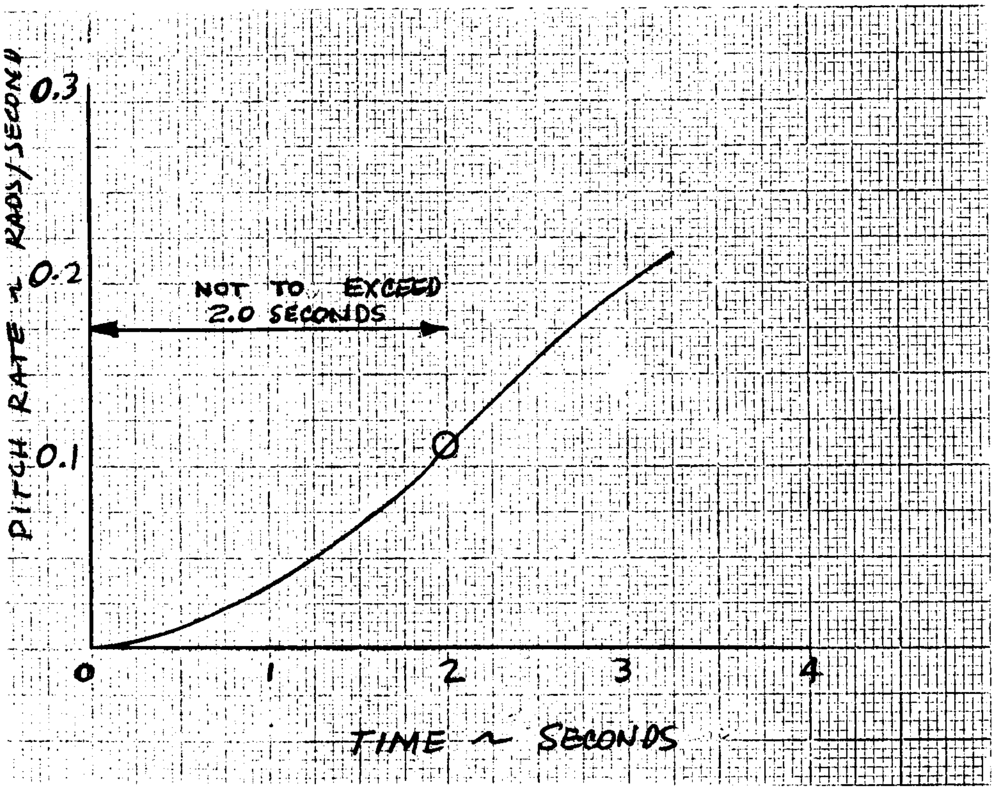


FIGURE 7I-3. PITCH ANGULAR VELOCITY RESPONSE
HOVER AND TRANSITION

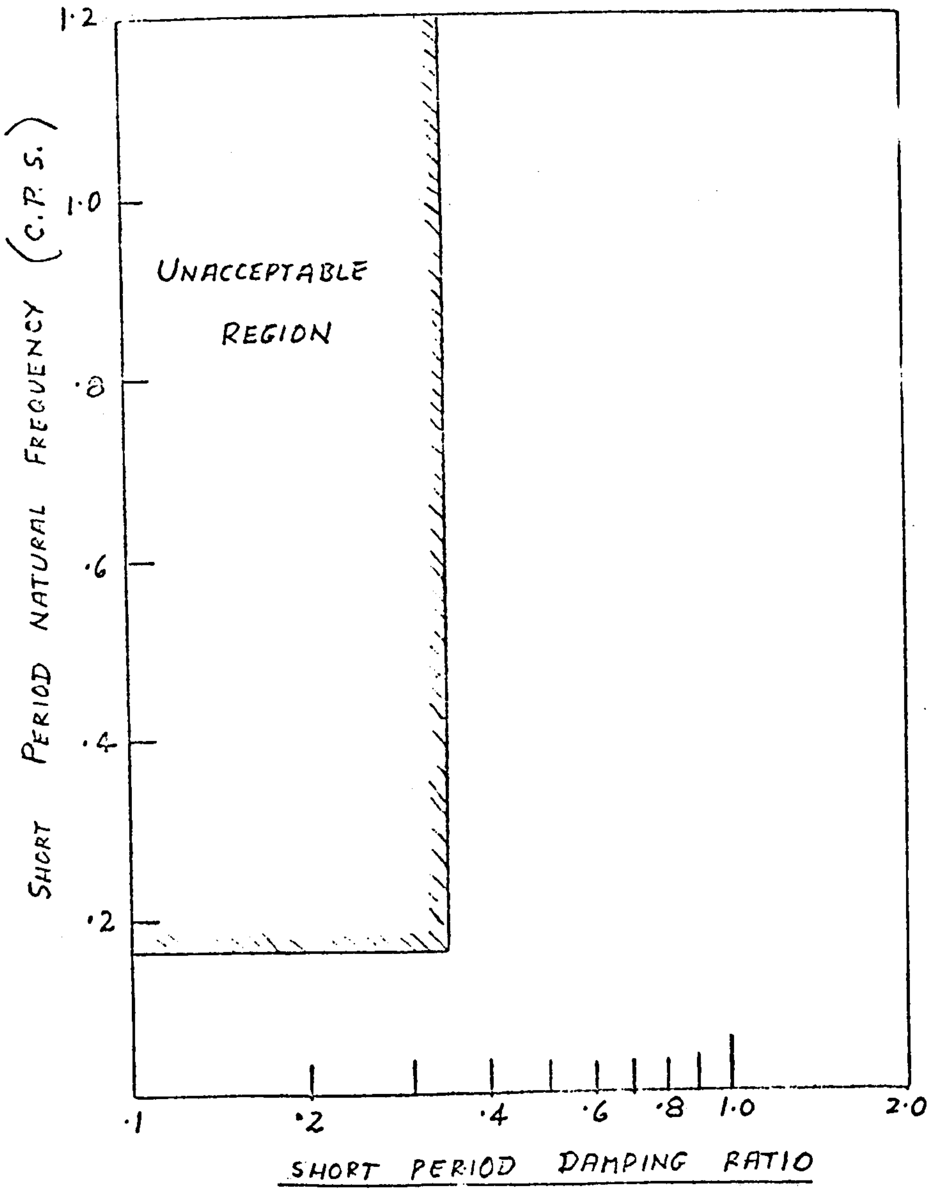


FIGURE 7I-4. LONGITUDINAL SHORT PERIOD DYNAMICS - MIL-F-8785 - CONVENTIONAL FLIGHT

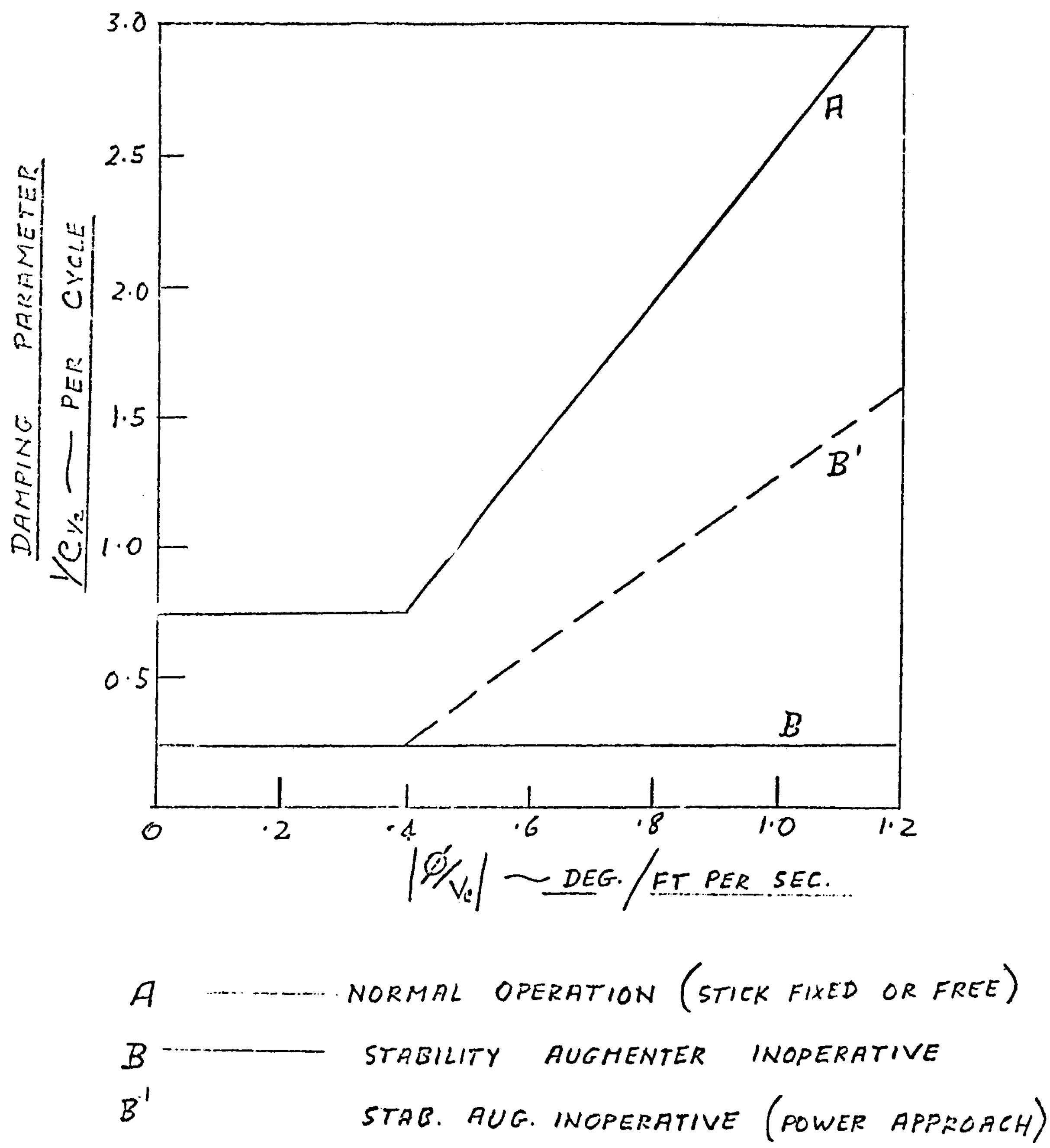


FIGURE 7I-5. LATERAL/DIRECTIONAL OSCILLATIONS - MIL-F-8785 - CONVENTIONAL FLIGHT

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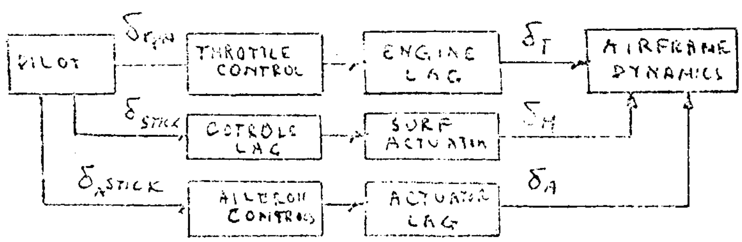
APPENDIX 7-II

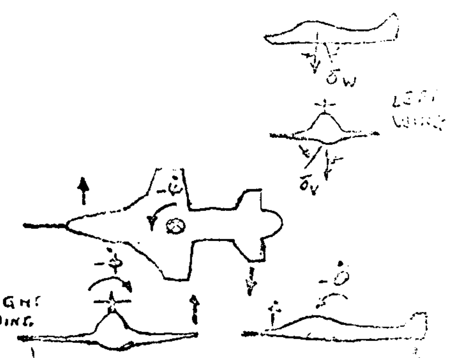
Desired: a) *Single Subscript
b) Denotes Output of Position Control
c) Positive Control Cause Negative Moment

Type	Symbol	Description	Polarity	Unit
Aero	δ_H	Horiz. Stabil. Defl.	Positive Trail. Edge Down	Deg
	δ_R	Rudder Defl.	Positive Trail. Edge Left	Deg
	δ_A	Aileron Defl.	Positive Right Ail. Down	Deg
Reaction Control	δ_Y	Pitch Reaction Jet Nozzle	Positive to Cause Increase Down Exhaust Stream in Aft Nozzle	Deg
	δ_Z	Yaw Reaction Jet Nozzle	Positive to Cause Increase Left Exhaust Stream in Aft Nozzle	Deg
	δ_X	Roll Reaction Jet Nozzle	Positive, to Cause Increase Right Wing Exhaust Stream Down	Deg
Engine Control	δ_T	Throttles	Positive for Thrust Incr.	Deg
	δ_W	Vectoring of Nozzle (Pitch)	Aft Defl. from Vert.	Deg
	δ_V	Vectoring of Nozzle (Roll)	Right Defl. from Vert.	Deg

*Second subscript denotes a distinction between two variables, such as in case of lift and lift/cruise throttle, δ_{TL}, δ_{TC}.



ILLUSTRATION



EXHAUST STREAM DIRECTION FOR POSITIVE CONTROL

FIGURE 7II-1. PROPOSED NOTATION FOR VTOL CONTROLS (BODY AXES)

LIST OF SYMBOLS

X, Y, Z	Forces along respective body axes
$\dot{u}$, $\dot{v}$, $\dot{w}$	Accelerations along respective body axes
L, M, N	Moments about the body X, Y, Z axes
$\dot{p}$, $\dot{q}$, $\dot{r}$	Rotational acceleration about the body X, Y, Z axes
I_x, I_y, I_z	Moment of inertia about the respective body axes
m	aircraft mass ~ slug
δ_A, δ_H, δ_R	Aero Dynamic control deflection to cause moment about X, Y, Z
δ_X, δ_Y, δ_Z	Reaction control deflection about the three body axes
δ_T	Thrust control deflection
δ_{WL}	Vector angle lift engines
δ_{WC}	Vector angle lift/cruise engines
p (P_g)	Roll angular velocity (measured by rate gyros)
q (q_g)	Pitch angular velocity (measured by rate gyro)
r (r_g)	Yaw angular velocity (measured by rate gyro)
u	Forward component of velocity
w	Normal component of velocity
v	Side component of velocity
α, β	Angle of attack and angle of side slip
H	Absolute altitude
θ	Pitch attitude
Φ	Roll attitude
Ψ	Yaw attitude
$\gamma_1, \gamma_2, \gamma_3, \gamma_4$	Components of simulated gust velocity
U	Total forward velocity
γ	Flight path angle
U_g, V_g	Forward and side velocity as measured by doppler ground velocities

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APPENDIX 7-III

CONVERSION DYNAMICS CALCULATION

Digital computer programs were used to find the necessary transfer function and time response for the airplane during conversion. Transfer functions to diverter valve command were found for pitch, $\frac{A_{\theta\,to}}{\Delta}$, and for normal velocity $\frac{A_{wto}}{\Delta}$ by using the incremental moment, ΔM_o, and the incremental forces, ΔX_o and ΔZ_o, which were created by the input of the diverter valve. Transfer functions to lift engines shut off were found for pitch, $\frac{A_{\theta t2}}{\Delta}$, and for normal velocity, $\frac{A_{wt2}}{\Delta}$ by using the incremental moment, ΔM_2, and the incremental forces ΔX_2, and ΔZ_2, inputs which were created by shut off of the lift engines. In order to get a single transfer function, each pair of transfer functions were combined using two second delay, e^{-2S}. Time response of θ and α for stability augmenter system, SAS, OFF; and of θ for SAS ON was found using the Laplace Transform method.

a. INPUT DATA

t=o, Diverter valve	t=2 sec, lift engines shut off
ΔM_o = 6,750 ft-lb	ΔM_2 = -23,400 ft-lb
ΔX_o = 3,870 lb	ΔX_2 = -7,320 lb
ΔZ_o = 3,520 lb	ΔZ_2 = 10,270 lb

b. Flight condition - V=180 Kn, W=18,000 lb clean, Basic Aero Data, see Section 2.4.4.

c. Combination of transfer functions, step input, SAS OFF

$$\theta(s) = \frac{1}{S}\left[\frac{1}{(0.05S+1)}\left(\frac{A_{\theta to}}{\Delta}\right) + \frac{e^{-2S}}{(0.5S+1)}\,\frac{A_{\theta\,t2}}{\Delta}\right]$$

$$= \frac{1}{S\Delta}\left[\frac{A_{\theta to}}{0.05S+1} + \frac{A_{\theta\,t2}e^{-2S}}{0.5S+1}\right]$$

$$\alpha(s) = \frac{57.3}{SU\Delta}\left[\frac{A_{wto}}{0.05S+1} + \frac{A_{wt2}\,e^{-2S}}{0.5S+1}\right]$$

where:

S	–	Laplace operator
$\frac{1}{0.05S+1}$	–	Diverter valve lag
$\frac{1}{0.5S+1}$	–	Lift engine lag

Δ	—	Free A/F denominator	
$A_{\theta to}$	—	Theta numerator, diverter valve command	SEE FIGURE 7-III-2
$A_{\theta t2}$	—	Theta numerator, left engine command	
A_{wto}	—	W numerator, divert valve command	
A_{wt2}	—	w numerator lift engine command	
e^{-2S}	—	2 sec. delay, See Figure 7-III-1	
57.3/U	—	1/5.288	

d. Combination of transfer functions, step input, SAS ON

$$\theta(s) = \frac{ACA\theta}{S\Delta\Delta_c}\left[\frac{A_{\theta to}}{0.05S+1} + \frac{A_{\theta t2}\, e^{-2S}}{0.5S+1}\right]$$

where:

Δ_c — Augm. A/F denominator. See Figure 7-III-3

$\left.\begin{matrix} A \\ C \\ A_\theta \end{matrix}\right\}$ — See definition in Figure 7-III-3

Delay e^{-2S} can be approximated as

$$\frac{(S^2 - ds + E)\ (S - F)}{(S^2 + ds + E)\ (S + F)}$$

Where:

For 2 Sec Delay

$$a = \frac{3.68}{2};\quad a = 1.84$$
$$b = 0.953\ a;\ b = 1.75352$$
$$c = 1.26\ a;\ c = 2.3184$$
$$d = 2\ a;\ d = 3.68$$
$$E = a^2 + b^2;\ E = 6.4604324$$
$$F = c;\ F = 2.3184$$

Thus:

$$e^{-2S} = -\frac{S^3 - 5.9984S^2 + 14.992144S - 14.97786645}{S^3 + 5.9984S^2 + 14.992144S + 14.97786645}$$

FIGURE 7-III-1

$t = 0$, DINSET VALVE COMMAND

$$\frac{A_{\theta_{t_0}}}{\Delta} = \frac{8.676\,S^2 + 2.483\,S + 1.156}{\Delta} = \frac{16.8\left(\frac{S + .142 \pm j.336}{.365^2}\right)}{\Delta}$$

$$\frac{A_{w_{t_0}}}{\Delta} = \frac{6.297\,S^3 + 46.654\,S^2 + .575\,S + 1.051}{\Delta} = \frac{15.3\left(\frac{S + .00464 \pm j.150}{.150^2}\right)\left(\frac{S + 7.40}{7.40}\right)}{\Delta}$$

$t = 2$ sec, LIFT ENGINES SHUTOFF

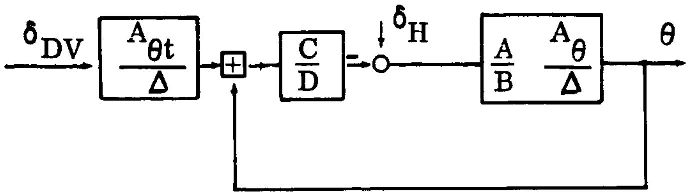
$$\frac{A_{\theta_{t_2}}}{\Delta} = \frac{-31.521\,S^2 - 31.840\,S - 2.540}{\Delta} = \frac{-32.0\left(\frac{S + .0873}{.0873}\right)\left(\frac{S + .923}{.923}\right)}{\Delta}$$

$$\frac{A_{w_{t_2}}}{\Delta} = \frac{18.37\,S^3 - 148.\,S^2 - 3.019\,S - 3.645}{\Delta} = \frac{-53.1\left(\frac{S + .0117 \pm j.156}{.156^2}\right)\left(\frac{S - 8.076}{8.076}\right)}{\Delta}$$

$$\Delta = 1.003\,S^4 + 1.278\,S^3 + 3.297\,S^2 + .0947\,S + .0687$$

$$= \left(\frac{S + .0105 \pm j.145}{.145^2}\right)\left(\frac{S + .626 \pm j1.69}{1.80^2}\right)$$

FIGURE 7-III-2. CONVERSION TRANSFER FUNCTION
(COMBINED MOMENT AND X AND Y THRUST)



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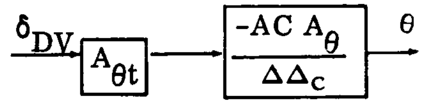

For $\delta_H = 0$ -stabilizer pilot input

$$\frac{\theta}{\delta_{DV}} = \frac{A_{\theta t}\,(-ACA_\theta)}{\Delta\Delta_c}$$

where:

$A_{\theta t}$ = $A_{\theta\,to}$ or $A_{\theta\,to}$ See Figure 6.2

A_θ = Free A/F Numerator to Stabilizer Input

A_θ = $-6.01553\ S^2 - 3.80334S - 0.183285$

A = 31.4

C = (0.061) (3943.84) (5S+1)

B = S + 31.4→Power Actuator Lag

D = $S^2 + 62.8S + 3943.84$→Servo Actuator Lag

Δ_c = $ACA_\theta + BD\Delta$→Augmented A/F Denominator

Δ_c = $1.0034S^7 + 95.797S^6 + 6059.5S^5 + 132120.S^4 + 404930.S^3 + 597940.S^2 + 47792.S + 9897.7$

Δ = $1.00339S^4 + 1.27759S^3 + 3.29688S^2 + 0.0947367S + 0.0687449$

FIGURE 7-III-3. SAS ON TRANSFER FUNCTION FOR DIVERTER VALVE OR LIFT ENGINES INPUT

e. List of total response functions for SAS OFF α (s), θ (s), and SAS ON θ (s). See Figure 7-III-4.

Inverse Laplace of these response functions was found and the result was plotted on Figure 7-III-5, 7-III-6, and 7-III-7.

Figure 7-III-5 shows airplane angle of attack, α plotted vs time, for SAS OFF. Figure 7-III-6 and Figure 7-III-7 show airplane pitch attitude θ, plotted vs time. Figure 7-III-6 shows airplane response for Stability Augmenter System off and Figure 7-III-7 for Stability Augmenter System ON.

d. To check the above results, step inputs were applied to diverter valve only, without turning the lift engines off. Figure 7-III-8 shows angle of attack response, SAS OFF. Figure 7-III-9 shows the pitch angle response, SAS ON.

$$\alpha(s) = \frac{57.3}{SU\Delta}\left[\frac{A_{\omega to}}{0.05S+1}\right]$$

$$\theta(s) = \frac{ACA_{\theta}}{S\Delta\Delta_c}\left[\frac{A_{\theta to}}{0.05S+1}\right]$$

Up to 2-seconds, Figure 7-III-8 is almost identical to Figure 7-III-5 and Figure 7-III-9 is very similar to Figure 7-III-6, which means that the approximation of the 2-second delay is good.

It should be noted, that Figure 7-III-9 represents SAS ON response. Unlike Attitude Hold mode, SAS was intended for stabilization of rate, and will not hold airplane attitude alone without pilot inputs.

1. SAS OFF

$$\theta(S) = \frac{1}{S\Delta}\left[\frac{A_{\theta to}}{.05S+1} + \frac{A_{\theta t2}\epsilon^{-2S}}{.5S+1}\right] = \frac{1}{S\Delta}\left[\frac{N_{um}}{D_{en}}\right]$$

$$= \frac{1}{S\Delta}\left[\frac{5.914305S^6 + 59.5999257S^5 - 15.4353305S^3 + 169.438724S^2 - 377.54936S - 20.728619}{.025S^5 + .69996S^4 + 4.6739236S^3 + 14.6185259S^2 + 23.22997S + 14.977866453}\right]$$

$$\alpha(S) = \frac{1}{5.288\Delta S}\left[\frac{2.23S^7 + 43.146S^6 + 471.9647S^5 - 261.8529S^4 + 3523.953S^3 - 1465.92S^2 + 38.9357S - 38.8416}{D_{en}}\right]$$

2. SAS ON

$$\theta(S) = \frac{-ACA_\theta}{\Delta\Delta_c S}\left[\frac{N_{um}}{D_{en}}\right]$$

where:

$$\Delta = 1.00339S^4 + 1.27759S^3 + 3.29688S^2 + .0947367S + .0687449$$

$$\Delta_c = 1.0034S^7 + 95.797S^6 + 6059.5S^5 + 132120.S^4 + 404930.S^3 + 597940.S^2 + 47792.S + 9897.7$$

$$A = 31.4$$

$$C = (.061)(3943.84)(5S + 1)$$

$$A_\theta = -6.01553S^2 - 3.80334S - .183285$$

FIGURE 7-III-4. AIRPLANE DYNAMICS DURING CONVERSION

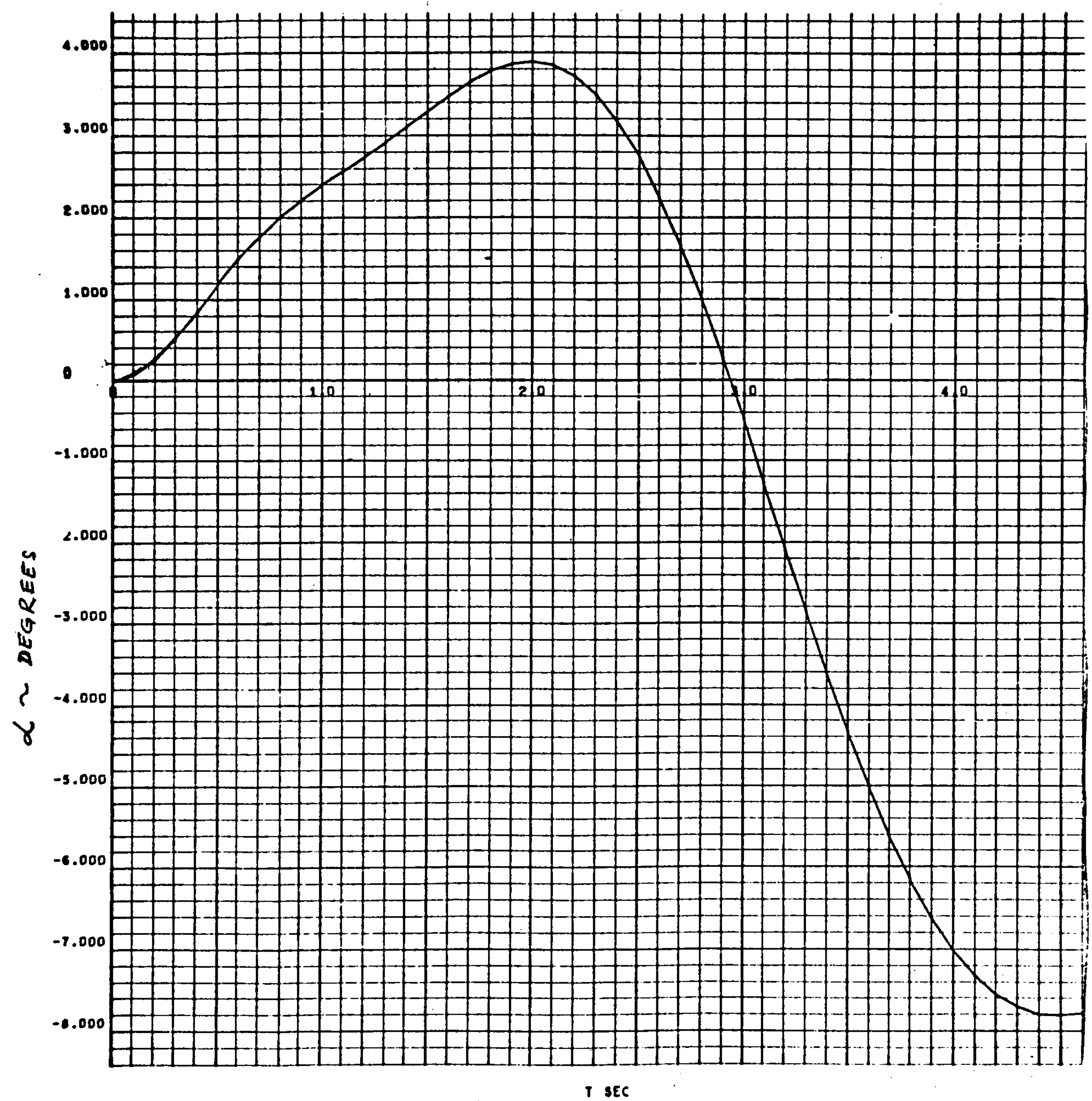

FIGURE 7III-5. CONVERSION DYNAMICS$_\alpha$, SAS OFF

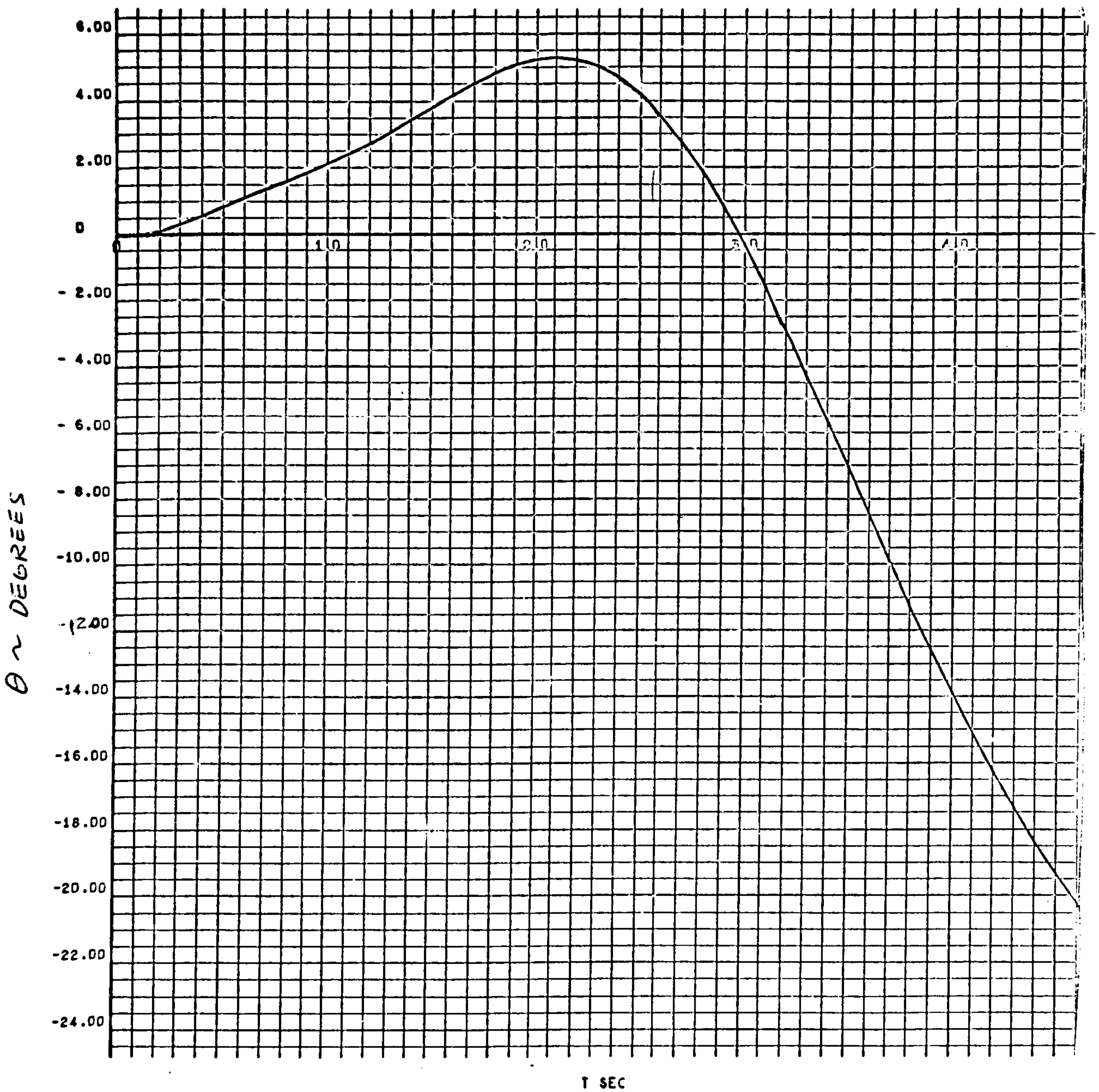

FIGURE 7III-6. CONVERSION DYNAMICS θ, SAS OFF

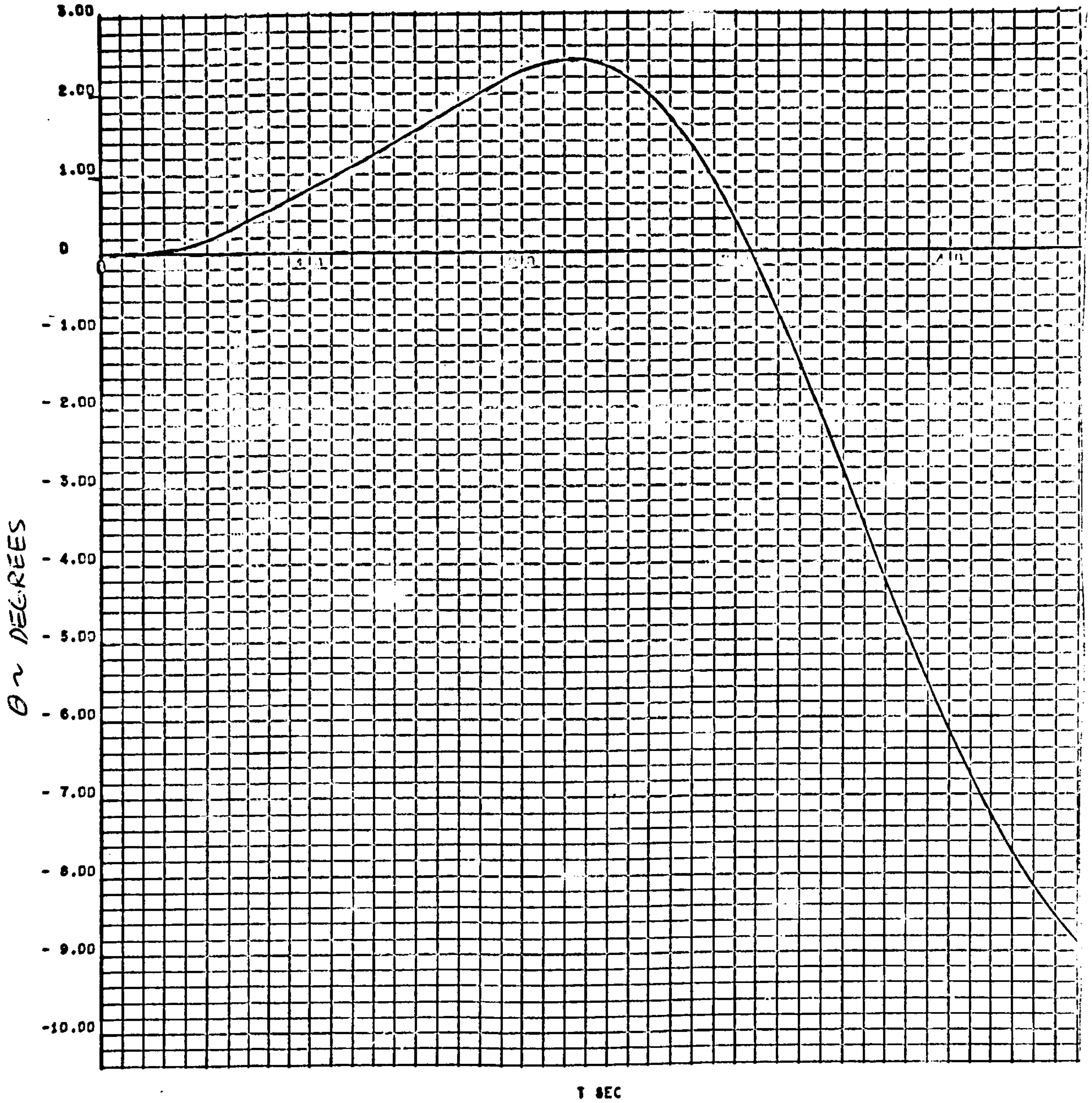

FIGURE 7III-7. CONVERSION DYNAMICS θ, SAS ON

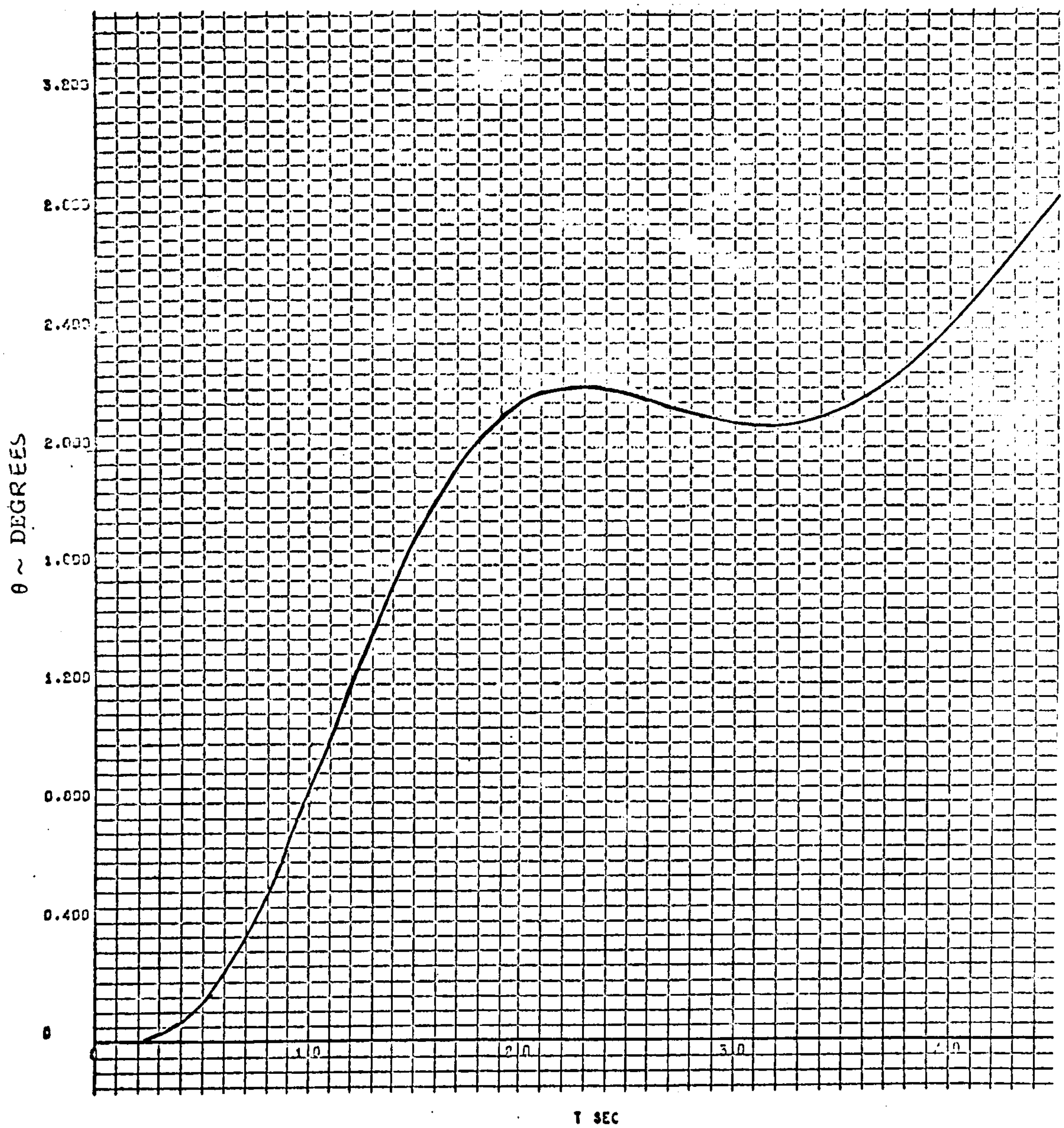

FIGURE 7III-8. AIRPLANE RESPONSE TO DIVERT VALVE STEP INPUT θ, SAS ON

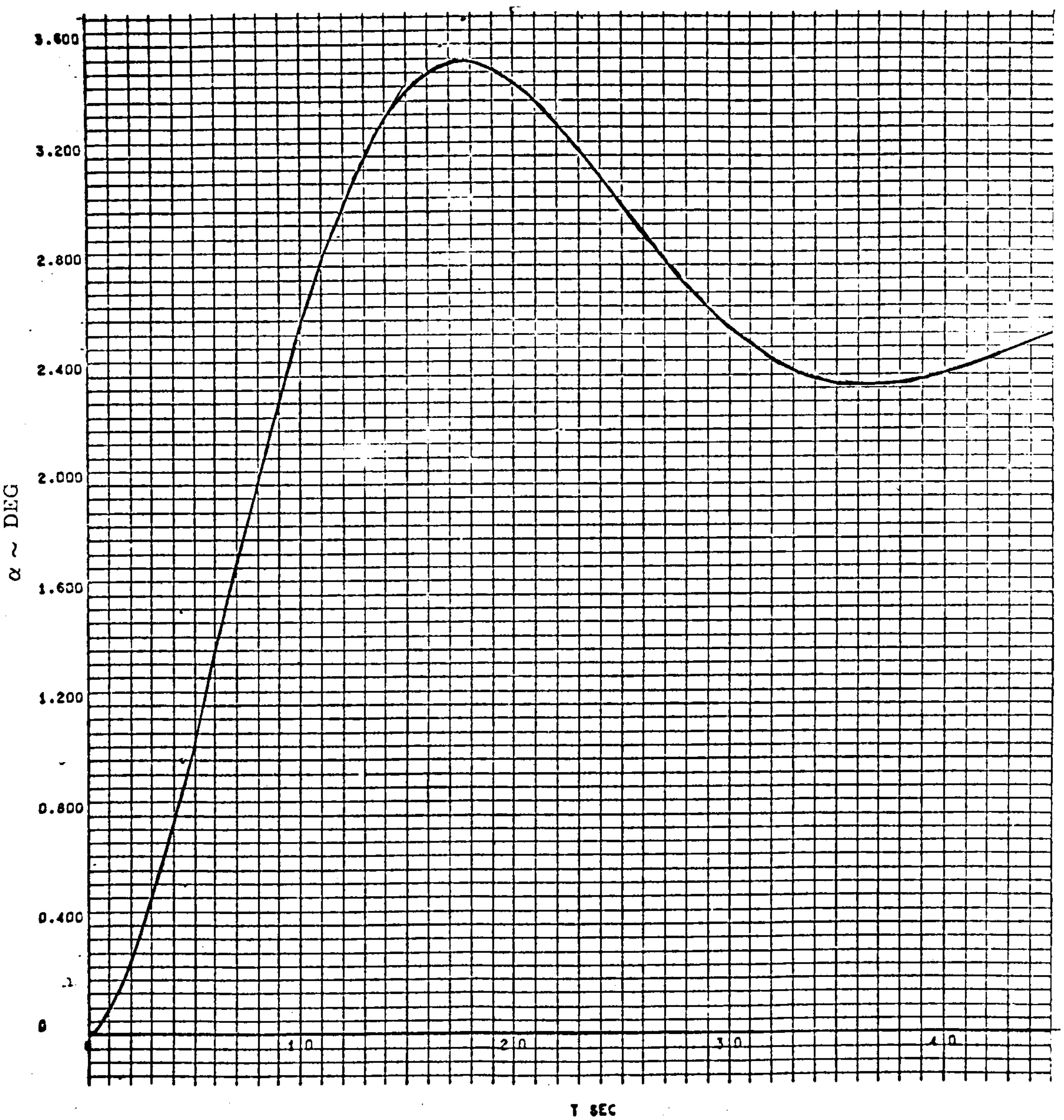

FIGURE 7III-9. AIRPLANE RESPONSE TO DIVERT VALVE STEP INPUT α, SAS OFF

8.0 RELIABILITY REQUIREMENTS

8.1 INITIAL RELIABILITY STUDIES

Primary subsystem reliability requirements for the V/STOL airplane will be established as required to assure a predicted safety reliability of not less than 0.999 for the 100 hour flight program. Safety reliability is the probability of experiencing no crew fatality or injury as a result of equipment failure. The analysis below assumes that 50 percent of the flight time is in the hover and transition modes. Reliability estimates are therefore based on 50 hours of hover and transition flight.

Six basic subsystems have been considered to be prime contributors to potential hazards in the hover and transition modes.

(1) Propulsion

(2) Flight Control (including hover attitude control)

(3) Electrical

(4) Fuel

(5) Landing Gear

(6) Hydraulic

For these subsystems, only those failure modes are considered which could necessitate escape from the airplane. These systems are backed up by the escape system for purposes of aircrew safety.

A reliability diagram for these critical subsystems is shown in Figure 8-1. Personnel parachutes are considered as GFE and are not included in the escape subsystem analysis.

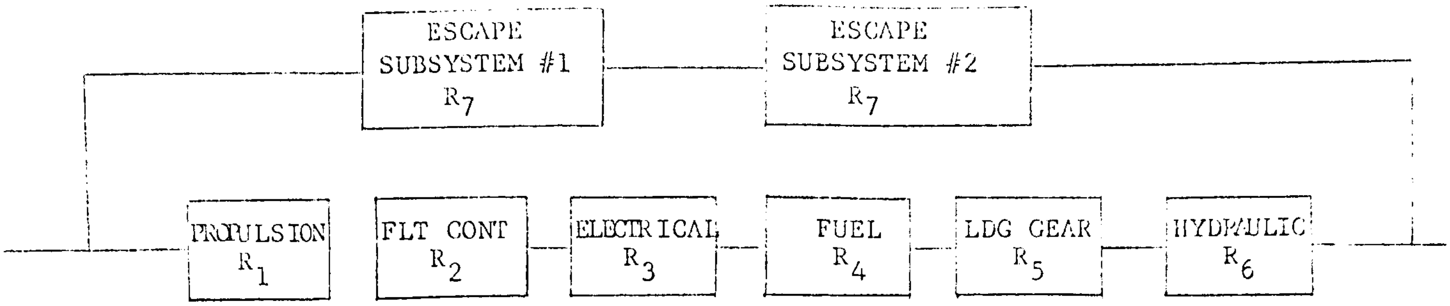

FIGURE 8.1 RELIABILITY DIAGRAM - SAFETY CRITICAL EQUIPMENT

R_1 through R_6 represent the probabilities of no failure for the respective subsystems which would require ejection in a single flight. R_7 is the probability of successful ejection for each pilot in the event of such failure.

Safety reliability may then be computed as follows for a two-airplane flight program.

Probability of no failures requiring ejection in a single flight is thus:

$$P = R_1R_2R_3R_4R_5R_6$$

Probability of no failure requiring ejection in 400 flights is:

$$P(O) = [P]^{400}$$

Probability of exactly one failure requiring ejection in 400 flights is:

$$P(1) = 400[P]^{399}[1 - P]$$

Probability of exactly two failures requiring ejection in 400 flights is:

$$P(2) = 1 - 400[P]^{399} + 399[P]^{400}$$

Safety reliability is then:

$$P(S) = 1 - [P(1)][1 - R_7^{\,2}] - [P(2)][1 - R_7^{\,4}]$$

For the five configurations initially considered, Tables 8-1 and 8-2 summarize the reliability estimates for the NASA flight test program for the composite and pure lift modes. Since hover and transition are considered to be the highest risk regimes, subsystem reliability levels adequate to assure achievement of the predicted safety reliability level of 0.999 for the hover and transition mode will be considered adequate for the entire program. Therefore, the reliability estimates are based on

50 hours in the hover and transition mode. Subsystems other than propulsion were considered essentially the same for the two configurations; therefore, only the propulsion subsystem was listed separately. Estimates of probabilities that the other critical subsystems will not fail in such a way as to necessitate ejection during the test program are as follows. These probabilities account for redundancies within the subsystems.

R_2 (Flight Control) = .99202

R_3 (Electrical) = .999992

R_4 (Fuel) = .99202

R_5 (Landing Gear) = .99960

R_6 (Hydraulic) = .999975

CONFIG.	R_P	$P_{(0)}$	$P_{(1)}$	$P_{(2)}$	$P_{(S)}$
N-309	.99963	.98303	.01681	.00016	.99932
MOD. T-39A	.99888	.98229	.01754	.00017	.99929

TABLE 8-1. RELIABILITY ESTIMATES - COMPOSITE MODE

CONFIG.	R_P	$P_{(0)}$	$P_{(1)}$	$P_{(2)}$	$P_{(S)}$
N-309	.99975	.98315	.01669	.00016	.99933
MOD. T-39A	.99888	.98229	.01754	.00017	.99929

TABLE 8-2. RELIABILITY ESTIMATES - DIRECT LIFT MODE

The following are definitions of the calculated parameters.

R_P = Probability of no propulsion failure requiring ejection during the test program.

$P_{(0)}$ = Probability of no failure requiring ejection during the test program.

$P_{(1)}$ = Probability of one failure requiring ejection (loss of one aircraft).

$P_{(2)}$ = Probability of two failures requiring ejection (loss of two aircraft).

$P_{(S)}$ = Safety reliability - probability of no fatality during the total test program (assumes a 0.98 reliability of each ejection seat).

It can be seen that while differences do exist between the estimates of $P_{(S)}$ for the two configurations, these differences are small enough that no significant choice can be made on the basis of reliability alone. Both configurations considered here exceed the 0.999 safety reliability ($P_{(S)}$) established as acceptable.

The calculations make the following assumptions:

1. Failure of one engine may be tolerated.
2. Alternate landing gear release is present.
3. Subsystem failures other than those considered in the analysis may be counteracted by flight abort.
4. Dual instruments are available for all required safety monitor functions.
5. Dual electrical and hydraulic systems are available.
6. The mean-time-between-failure of the engines is equal to 500 hours.

Using these assumptions, preliminary reliability prediction analysis performed indicate that the assigned subsystem reliability goals can be met by existing state-of-art hardware.

The probability of successful ejection used in the analysis above is predicated on independence from other failures. Specifically, attainment of this value depends on the pilot not being subjected to excessive sink rates or rotational rates in the event of failure of critical systems. This problem involves the flight control system components; therefore fail-safe requirements must be specified in those areas where failure could result in a hard-over condition of the controls.

8.2 EQUIPMENT RELIABILITY REQUIREMENTS

The equipment reliability summary shown in Table 8-3 indicates the state of development of each subsystem defined as safety critical (those whose failure could result in the loss of an aircraft or personnel injury). The MTBF's listed as V/STOL requirements are mean-time-between-failures that would result in loss of the aircraft or personnel injury. These MTBF's are consistent with the reliabilities developed in paragraph 8.1 to meet the overall safety reliability of 0.999 established as acceptable. For completeness, some equipment items are listed even though at this time they are considered as non-critical; for example VSS failure would not cause loss of the aircraft, assuming that the safety pilot can switch it out of the system. This summary will be updated during the development phase to include information not now available, and to include a more detailed subsystem breakdown where applicable.

Reliability estimates for the two candidate configurations are summarized below. These are based on the assumption that 50 percent of the NASA flight test program (50 hours) will be spent in the hover and transition mode. Subsystems other than propulsion were considered essentially the same for each configuration; therefore, the only difference in reliability is due to the number of engines used. These probabilities are for the entire 100 hour test program.

Configuration	R_P (Propulsion)	$P_{(0)}$	$P_{(1)}$	$P_{(2)}$	$P_{(S)}$
MOD. T-39A	.99888	.98229	.01754	.00017	.99929
N-309	.99910	.98249	.01734	.00017	.99930

TABLE 8-3. RELIABILITY ESTIMATES

R_P = Propulsion reliability (allows one engine out).

$P_{(0)}, P_{(1)}, P_{(2)}$ = Probability of zero, one or two failures requiring ejection (loss of zero, one, or two aircraft).

P_S = Safety reliability (no crew fatality as a result of equipment failure).

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